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1100 SEVENTEENTH STREET, N.W. WASHINGTON, D.C. 20036

COVER SHEET FOR TECHNICAL MEMORANDUM

TITLE- Mars Excursion Module Ascent Propulsion Stage Design

TM- 68-1013-3

DATE- July 8, 1968

FILING CASE NO(S)- 730

AUTHOR(S)- M. H. Skeer

FILING SUBJECT(S)- (ASSIGNED BY AUTHOR(S)- Mars Manned Landing Propulsion Vehicle

ABSTRACT

A previous memorandum by the author considered the design of a minimum Mars Excursion Module (MEM) ascent astronaut capsule in which required payloads comprising crew, science payload, subsystems and structure were derived. Approximately 700 lbs and 1,300 lbs were estimated for one and two man ascent capsules, respectively. In this study, design of a MEM ascent propulsion vehicle to deliver the required payloads to orbit from the surface of Mars is undertaken with the prime purpose of identifying fruitful areas of technological research and development needed for evaluation and support of future program planning options. The study includes designs of both earth storable (Compound A/MHF-5) and space storable (FLOX/Methane) propulsion systems sized for return to a highly elliptical Mars parking orbit.

While the earth storables offer superior handling characteristics, the space storable propellants offer lower vehicle weight and single stage ascent, versus two stages for the earth storables. The analyses indicate no serious problems associated with storing space storable propellants if refrigeration is available in transit to Mars. At least two weeks storage on the surface of Mars is possible without active refrigeration.

Commonality of the ascent propulsion vehicle with the unmanned Mars surface sample return (MSSR) mission is briefly assessed and found quite attractive.

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COVER SHEET FOR TECHNICAL MEMORANDUM

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(NASA-CR-97688) MARS EXCURSION MODULE ASCENT PROPULSION STAGE DESIGN (Bellcomm, Inc.) 56 p

TITLE- Mars Excursion Module Ascent Propulsion Stage Design

TM- 68-1013-3

DATE- July 8, 1968

FILING CASE NO(S)- 730

AUTHOR(S)- M. H. Skeer

FILING SUBJECT(S)- Mars Manned Landing
(ASSIGNED BY AUTHOR(S)- Propulsion Vehicle

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While the earth storables offer superior handling characteristics, the space storable propellants offer lower vehicle weight and single stage ascent, versus two stages for the earth storables. The analyses indicate no serious problems associated with storing space storable propellants if refrigeration is available in transit to Mars. At least two weeks storage on the surface of Mars is possible without active refrigeration.

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1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Mars Excursion Module Ascent Propulsion Stage Design - Case 730

DATE: July 8, 1968

FROM: M. H. Skeer

TM: 68-1013-3

TECHNICAL MEMORANDUM

1.0 INTRODUCTION

The achievement of minimum Mars Excursion Module (MEM) gross weight, as suggested by sensitivity factor studies, is contingent upon minimizing ascent (Mars surface to orbit) vehicle weight. When possible, therefore, subsystems should be located in the descent stage, and operational modes in association with the parent spacecraft judiciously selected to place weight penalties on the spacecraft, rather than on the ascent vehicle itself.

A previous memorandum (Reference 1) considered the design of a MEM ascent astronaut capsule, which fully utilized this conceptual approach (Figures 1 and 2). Required payloads (comprising crew, science payload, subsystems, and structure) of approximately 700 lbs and 1,300 lbs were estimated for one and two man vehicles, respectively. (Alternately, the larger capsule can be considered as a one man vehicle with substantially greater payload capability.) In the present study, design of MEM ascent propulsion vehicles to accommodate the previously derived payloads is undertaken with the prime purpose of identifying fruitful areas of technological research and development needed for evaluation and support of future program studies.*

The scope of the study undertakes the design of both earth storable (Compound A/MHF-5) and space storable (FLOX/Methane) propulsion systems sized for Mars surface return to a highly elliptical 24 to 48 hr Mars parking orbit.

The preliminary portion of this memorandum describes a mission profile from which general ground rules are implicitly derived. In the ensuing sections alternative design and configuration concepts are considered and subsystems evaluated

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*Since the astronaut uses the ascent capsule only during descent and ascent (see Section 2.0) the ascent capsule weight is basically independent of surface staytime. Therefore, ascent and landed payloads may be considered essentially decoupled; and descent payload may be sized independently for varying staytime and mission requirements. In this context ascent systems do not impose a fixed mission payload capability.

on an item by item basis. Propellant storage constraints imposed by the wide gamut of mission environments are explored in detail.

2.0 MISSION PROFILE

The MEM arrives in the vicinity of Mars with an Orbiter mission module which, via retropropulsion or aerodynamic braking, establishes a highly elliptical (24 to 48 hour) capture orbit with periplanet velocity slightly below escape, i.e., at about 16,000 fps. This orbit is non-optimum for MEM ascent, but is desirable to minimize main module capture and escape velocities. The MEM Separates from the parent ship and descends to the surface either by direct entry (prior to the capture maneuver) or from elliptical orbit (Figure 3) by aerodynamic braking and retropropulsion (Figure 4). An arbitrary staytime, perhaps one week to 30 days, is provided during which time surface reconnaissance and experiments are performed. The astronauts return in the ascent stage and rendezvous with the parent module in elliptical orbit.

Abort capability is provided prior to MEM entry, for a period of time shortly before touchdown and from the surface in the event of surface shelter failure.

The MEM descent vehicle is a cone or Apollo shaped entry shell which contains heat shield, retropropulsion, landing gear, the ascent vehicle, and perhaps a laboratory and shelter for surface operations. Alternately, the latter two items can be delivered in a separate vehicle. The ascent vehicle houses descent command system control interfaces, the ascent capsule, and ascent propulsion stages. Abort on entry necessitates that the astronauts ride in the ascent stage capsule to allow rapid escape.

The relatively heavy entry landing systems (i.e., computers, guidance and communications subsystems) are packaged in the descent stage and connected to the ascent/command capsule by umbilicles (or an inductance couple) capable of being broken immediately in case of abort launch.

Surface and abort launch (Figure 5) are achieved via preprogrammed trajectories to low circular orbit. A single orbit coast (or less) is allowed for positioning and orbit determination from the main module, thus requiring ascent vehicle engine restart capability. Transfer is achieved so that the MEM ascent stage (MEM/AS) is slightly ahead of the parent spacecraft. The MEM/AS is guided by radio command from the main module during the final phases of the rendezvous sequence. At rendezvous, the astronaut either flies the MEM/AS into a prepared docking area, or leaves the spacecraft and maneuvers to the main module by EVA. The nominal mission then requires an astronaut to live in the ascent stage for perhaps 6 hours before landing and 1 or 2 hours after ascent.

3.0 CONCLUSIONS AND SUMMARY OF PRINCIPAL FINDINGS

With mission profiles in perspective, the principal results of this study are summarized below:

- SPACE STORABLE PROPELLANTS RECOMMENDED

  Earth storable propellants have obvious advantages of ease of handling and storage; however, elimination of a second stage and reduced weight are strong factors favoring a space storable system. From a feasibility standpoint, space storables are considered competitive with earth storables for future MEM mission applications.

- PROPELLANT REFRIGERATION ADVANTAGEOUS

  Utilization of active spacecraft refrigeration is fundamental to efficient design of the small FLOX/Methane stages. It is decidedly within the capability of currently achievable lightweight refrigeration systems to provide thermal control throughout the entire MEM environmental spectrum (i.e., prelaunch packaging, earth storage, earth orbital, pre-injection, trans Mars, and Mars surface storage). Pre-Mars-entry subcooling enables passive storage on the surface to be achieved for up to two weeks staytime. For indefinite staytimes, the active refrigeration penalty is estimated to be approximately 250 lbs (refrigerator plus power unit).

- Gross weight of <u>two-stage</u> vehicles employing Compound A/MHF-5 propellants are approximately 5,300 lbs and 8,900 lbs for one and two man payloads, respectively.

- Weights of <u>single stage</u> FLOX/Methane vehicles are correspondingly 4,170 lbs and 7,420 lbs.

- SINGLE STAGE ASCENT VEHICLE RECOMMENDED

  A single stage space storable vehicle is weight competitive with a two stage space storable vehicle, and moreover, offers substantially improved packaging arrangements.

- The two man earth storable and space storable vehicles afford 16% and 11% weight savings per man compared to respective one man vehicles. In both cases 7% is attributed to reduced capsule weight, and the remainder from improved stage performance.

- Compound A/MHF-5 and FLOX/Methane propellants have special thermal characteristics which make them suitable for greatly simplified handling and storage techniques when active thermal control is utilized. By judicious selection of common fuel/oxidizer storage temperatures, the propulsion stage in both cases can be prepackaged prior to launch and maintained without propellant transfer or venting for the duration of the mission--with little mass fraction penalty.

- MSSR/MEM commonality is entirely reasonable for all multi-planet flyby missions. For the high velocity Mars twilight flybys, addition of another propulsion stage is sufficient.

4.0 VELOCITY/THRUST ESTIMATION

4.1 Velocity Determination

Required velocities for a class of highly elliptical Mars capture orbits are given in Figure 6. Total velocity including gravity, drag, and aerodynamic losses were determined in Reference 3 for a "near optimum" trajectory ellipse. Total impulsion velocity, including drag and gravity losses, is estimated for a 23 1/2 hour (one day) ellipse to be approximately 18,600 fps. This analysis assumed drag characteristics for the single stage vehicle given in Figure 7 (Reference 4). An additional 400 fps is allowed for course correction resulting in a design nominal of 19,000 fps.

Optimum staging velocities were obtained from the perturbation formula in Reference 5, taking into account variable specific impulse and mass fraction effects.

4.2 Thrust/Weight (T/W) Determination

Initial thrust to weight (Reference 3) to achieve a 100 NM circular parking orbit was determined to be near optimum at 1 earth g initial acceleration, for first and second stages. Variable thrust was not considered. A 90 second coast gravity turn after first stage separation is required to prevent parking orbit overshoot. It is probable that direct rendezvous to elliptical orbit would allow higher T/W to achieve several hundred fps reduction in total velocity.

5.0 DESIGN CONCEPTS AND CONFIGURATIONS

The point designs considered in this study (six in all) are tabulated below. Only two stage vehicles are considered in the earth storable case. Both one and two stage vehicles are considered for space storable designs.

Propellant Class	Earth Storable		Space Storable			
Propellant	(Compound A/MHF-5)		(FLOX/Methane)			
Payload Class	700 lbs	1,300 lbs	700 lbs		1,3000 lbs	
Number of Stages	2	2	1	2	1	2

STAGE DESIGN SUMMARY

In spite of operational difficulties associated with handling and storage, the high performance of space storable propellants, which allows elimination of a complete propulsion stage, warrants their prime consideration as a design alternative to the earth storable stages.

Each of these concepts is developed in ensuing discussions.

5.1 Capsule/Stage Integration

Physiological constraints of manned entry govern the manned capsule configuration, which is relatively large compared to the ascent propulsion vehicles. As a consequence, unique integration situations result.

The one man capsule (Figure 1) is approximately a half cylinder sized to accommodate a suited astronaut in supine high G entry posture. The astronaut is supported in a net cradle which distributes entry loads to a structural frame forming the capsule perimeter (the entry g load is the predominant structural design condition). The capsule cover is a lightweight unpressurized shroud which serves principally to minimize drag losses and provides a passive thermal barrier during descent and ascent. Life support is contained entirely within the suit loop.

The two man capsule has a square frontal area formed by two rectangular couch frames similar to the construction of the one man design. A circular aerodynamic shroud is provided to minimize drag losses; the aft end of the shroud is sculptured to fit the frame periphery.

5.2 Earth Storable Vehicle Configurations

The one man vehicle configuration is shown in Figure 8. Propellant is packaged in a double barrelled cylinder arrangement contained completely within the payload shadow. This is to minimize aerodynamic drag losses; significant, even in the tenuous Martian atmosphere.

Due to CG control requirements, the fuel and oxidizer tanks must be placed in a fore and aft arrangement (as oppossed to side by side). The case illustrated shows two tanks with a common bulkhead to separate fuel and oxidizer. Compound A/MHF-5 share an extensive common liquidous range and may be suitably stored in this mode.

Stage 1 thrust loads are transmitted through the propellant tankage by means of a contoured thrust structure which distributes loads uniformly along a peripheral arc to the tank skins. The thrust loads are transmitted directly to the capsule frame via central column and peripheral tank extensions. Engine thrust loads are so small (ascent thrust/weight is 1 earth g) that a minimal interstage structure is adequate for ascent requirements. The second stage engine is mounted directly to the payload frame. Propellant containers are suspended from the frame by struts.

Mars entry and descent loads are limited to a maximum of 10 g's or less, consistent with manned entry allowances. The stage is supported at the base of the stage 1 propellant containers by extended circumferential skirts.

The two man capsule (Figure 9) has approximately double the frontal area of the one man vehicle, so separate fuel and oxidizer tanks can be utilized without increasing the vehicle frontal area. The four tanks are supported by a central ring which distributed loads uniformly to the tanks along a peripheral arc. Thrust loads are directed through the tanks to a hard point at a central frame which also serves as the couch support.

5.3 Space Storable Vehicle Configurations

Space storables are currently considered practical for application to large stages where long term passive storage can be achieved. MEM propellant volume is however insufficient to make passive storage feasible without accruing significant operational and weight penalties. Here, however, the opposite is the case. Propellant volume is <u>so</u> small that active thermal control can be employed to considerable advantage. Potential benefits derived from this approach are

- Achievement of mass fractions comparable to the earth storable stages,
- Improved packaging configurations,
- Elimination of boil-off penalties and spacecraft venting requirements,
- Avoidance of onboard propellant transfer,

- Ease of handling and loading,
- Open-ended Mars, earth and trans Mars storage capability, and
- Up to several weeks non-vented passive storage on Mars by subcooling prior to entry.

The entire vehicle, excluding the manned capsule, is stored in an insulated container. Internal temperature is actively maintained within the common liquidous range of the propellants. The insulation container is a "separator" rather than a pressure vessel so that no significant pressure differential exists across the insulation walls. Prior to Mars entry the stage is subcooled and the container is purged with an inert gas such as N_2 or argon to prevent CO_2 entry and subsequent condensation on the tanks on entry to Mars. If required, the purge can be maintained during surface staytime by means of N_2 stored on the descent stage.

The configurations of the two stage space storable vehicles (Figures 10 and 11) are, in general, similar to the earth storable vehicle designs. Exceptions are the manned cabin/stage 2 interface, and the stage 1/support interface. Here suitable insulation materials and low heat transfer supports are employed.

Single stage vehicle configurations (Figures 12 and 13) differ decidedly from the two-stage designs. The one man vehicle is a multi-cell container formed by three intersecting cylinders. Engine thrust loads are carried through the tanks at the four cylinder intersection points to the capsule frame. Separate fuel/oxidizer tanks or common bulkhead tankage both offer suitable packaging arrangements.

The two man vehicle is formed by ellipsoid oxidizer tank and a toroid or sphere-toroid fuel tank. Thrust loads are carried by a cone developed as a continuation of the oxidizer tank. The fuel tank is supported by the thrust cone and a cylindrical shroud at the tank perimeter. The manned capsule is supported at discrete points at the forward end of the cylinder.

6.0 PROPELLANT STORAGE CONSIDERATIONS

The stage designs must accommodate unique thermal control requirements imposed by the spectrum of mission environments including:

- Prelaunch packaging and earth storage,

- Post launch earth parking orbit,
- Trans Mars, and
- Mars entry and surface staytime.

For reference, Martian diurnal surface temperature variations, as given in Reference 6, are summarized below:

Condition	Temperature
Maximum	80°F
Mean Day Side	10°F
Mean Night Side	-100°F
Mean Planet	-45°F
Mean Amplitude of diurnal variation	110°F

6.1 Prelaunch Packaging and Storage

Before considering thermal storage requirements the packaging sequence is briefly addressed to illuminate problems associated with this procedure.

Propellant loading may be accomplished either prior to launch assembly (prepackaged), on the pad, or by propellant transfer in space. The latter two techniques employ LN_2 chill-down and purge (for FLOX/Methane), and propellant transfer from a common reservoir. By dint of the number of fill operations required* (with associated plumbing complexity) and inherent safety problems of onboard transfer, ground loading is strongly preferred. In subsequent discussion prepackaging is presumed. Advantages of the prepackaged mode are complete assembly and checkout prior to pad assembly, and elimination of tank venting except at initial loading.

6.2 Prepackaging Thermal Control Requirements

Compound A/MHF-5 and FLOX/Methane propellants each have special thermal characterisitcs which make them suitable

*The ascent stage is emplaced in the descent stage, and the entire vehicle installed in a probe hangar module. As many as three MEM vehicles, in addition to unmanned probes, are stored in the hangar (i.e., two manned vehicles and a separate shelter vehicle). In total 2 or 3 vehicles, with 3 to 7 separate stages are utilized.

for greatly simplified handling and storage techniques when active thermal control is utilized. By judicious selection of a common fuel/oxidizer storage temperature, the stage can be prepackaged prior to launch and in both cases, maintained without propellant transfer or venting for the duration of the mission with no mass fraction penalty. The means of achieving this is rather involved and coupled to four basic factors:

- Maximum oxidizer vapor pressure,
- Minimum fuel vapor pressure,
- Minimum gage pressure vessel design, and
- Storage/operation pressure duty cycle.

The concept developed in ensuing discussion is 1) to establish maximum allowable oxidizer vapor pressure by the minimum gage propellant container pressures, and 2) establish minimum ground and launch fuel vapor pressure at (or slightly below) 14.7 psia to prevent fuel tank buckling. The limiting operating (engine on) oxidizer vapor pressure is fixed by maximum operating vapor pressure, less the margin afforded by net positive suction head (NPSH), and acceleration and line losses which are not additive in the storage condition. Upon achieving a vacuum condition in orbit the 14.7 psia fuel tank pressure requirement is relaxed, allowing a subsequent reduction in storage temperature.

Impact of the various environmental conditions on storage requirements are considered in turn below. Additional factors such as minimum gage pressure vessel constraints and duty cycle pressure are unified with this discussion in section 9.2.1, ultimately resulting in selection of pressure vessel design criteria.

6.3 Compound A/MHF-5 Storage Considerations

Prelaunch Packaging and Earth Storage - Maximum storage temperature of Compound A/MHF is effectively limited by Compound A vapor pressure buildup. Minimum storage temperature is governed by low MHF-5 vapor pressure resulting in fuel tank compression in the 14.7 psi environment. Coincidently the optimum propellant storage temperature, striking a balance between these two constraints, is a convenient -77°F. A 2° excursion deadband, (i.e., 75°F to 79°F) is arbitrarily selected to estimate pressure vessel weight penalties. The temperature deadband requirement is a standard handling procedure for many classes of missile systems and should not pose operational difficulties. Active thermal control for earth storables is required only through launch.

Space Environment - in the absence of an external 14.7 psi atmosphere pressure the stage can be passively stored at any convenient temperature within the common liquidous range of 79°F to -70°F, the lower limit being the feezing temperature of MHF-5.

Mars Surface Environment - A potential surface storage problem is MHF-5 partial freezing. This can be avoided by either of several passive thermal control techniques:

1) Storing the stage at mean surface equilibrium temperature (-45°F) and limiting propellant temperature excursion by tank insulation to greater than -70°F, or

2) Raising the mean temperature of the entire vehicle by suitable thermal design (i.e., coatings, controlled "one way" heat leaks, etc.).

6.4 FLOX/Methane Storage Considerations

Prelaunch Packaging an Earth Storage - Baseline fill concept for the space storable propellants is prepackaging prior to pad assembly, with continuous active cooling maintained prior to launch by a refrigeration system incorporated in the hangar. Heat flux inputs on the surface are substantially greater than in space (by approximately an order of magnitude) because of degraded insulaton performance under atmospheric pressure. However, unlimited power for refrigeration is available and a favorable weight tradeoff results if the same insulation system is used.

Post Launch - The probe hangar is launched separately via Saturn V to high elliptical parking orbit and vented to obtain the benefit of vacuum to improve the insulation performance. Several months may elapse before docking and assembly of the probe hanger and the spacecraft is accomplished. In the interim (before spacecraft power is available) probe hangar and refrigeration power must be supplied by a power supply unit, which may be jettisoned before trans Mars injection. Alternately, because of low heat input in the vacuum space environment, propellants can be subcooled prior to launch and stored passively until spacecraft power is available. It is estimated that at least several months of passive storage can be achieved without need for refrigeration or venting if adequate subcool is provided.

Trans Mars - Upon trans planetary injection the launch and entry supports are disconnected. The stage is cooled by refrigeration, power being supplied by the main spacecraft. The stage is subcooled to minimum common liquidous range temperature prior to Mars entry.

Mars Entry and Surface Staytime - The MEM separates from the capsule several hours prior to entry. At separation, ascent/descent stage structural ties (which are disconnected for storage purposes) are re-established, active cooling is discontinued (presuming a short surface staytime), and the insulation compartment is pressurized with N_2, to Mars surface pressure.

During entry, heat flux can conceivably be increased by several orders of magnitude above the surface rate. However, because of the short duration, even with this conservative estimate, total heating input is quite small, raising propellant temperature by not more than several degrees.

Surface staytimes of up to several weeks are achieved by passive thermal control. The N_2 purge limits CO_2 icing (a nominal amount of which is tolerable). Longer staytimes require active cooling by a light weight refrigerator system utilizing solar panel or RTG power supply.

Prior to Mars surface launch, the insulation is separated from the ascent vehicle, the forward insulation remaining with the stage through ascent.

7.0 DESIGN SUMMARY

The vehicle characteristics derived for the classes of one and two man vehicles are given in Table 1. Included are estimated propellant performance and propellant mixture characteristics for varying engine thrust levels, initial stage thrust/weight, optimum staging velocities, and stage and vehicle gross weight. Stage and vehicle growth factors measure the sensitivity of vehicle weight to payload.

Sensitivity factors (S) given in Table 2 enable percentage gross weight changes (W) to be determined as a function of percentage change in mass fraction (λ) by the following formula:

$$\frac{\Delta W}{W} = [S_1 \frac{\Delta\lambda_1}{\lambda_1} + S_2 \frac{\Delta\lambda_2}{\lambda_2}]$$

Gross weights to elliptical orbit (19,000 fps) are approximately 5,220 lbs and 8,860 lbs for one man and two man earth storable stages, respectively.

Comparative FLOX/Methane vehicle weights are 4,090 lbs and 7,320 lbs for two stage vehicle and 4,170 lbs and 7,400 lbs for single stage vehicles. The single stage space storable is competitive weightwise with the two stage vehicle and, for operational and development simplicity, is strongly preferred.

The one and two man single stage space storable vehicles provide weight savings of 21% and 16% per man above respective two stage earth storable vehicles. This includes insulation penalty for short staytimes of less than 2 weeks. Extended staytimes result in a 200 lb to 300 lb refrigerator and power system penalty. In the latter case the net savings over earth storables is approximately 17% and 14% for one and two man stages, respectively.

Stage weight breakdowns are given in Table 3 and dimensional characteristics in Table 4. (Subsystems analysis upon which these weights are derived is presented in Section 9.) Calculations are based on currently achievable, or moderately advanced state-of-the-art technologies.

A single weight iteration was employed in mass fraction calculations. Finer estimates can be made directly by consideration of the sensitivity factors in Table 2.

8.0 PROPELLANT CHARACTERISTICS

8.1 Earth Storable Selection

The earth storable propellants selected for point design analysis are Compound A/MHF-5 (5/5) typical of the families of advanced chlorinated propellants currently under serious consideration for tactical weapons systems.

Development of 5/5 propellants has proceeded to the point where small scale static firings (less than 10,000 lbf thrust) have been made. 5/5 is currently candidate for the Condor missile funded under fixed price development (Reference 7). The Navy has accepted specifications for characterization of 5/5 as satisfactory.

Performance curves for 5/5 as a function of thrust are given in Figure 14 (Reference 8). These values are considered to be slightly optimistic but achievable. Typical I_{sp}'s are from 344 sec to 353 sec vacuum for mixture ratios (MR) of 2.7, chamber pressure (P_c) of 1,000 psia, and expansion ratio (ε) of 40.

Temperature characteristics relating to common liquidous range (CLR) are given below:

State Condition	Compound A	MHF_5
Freezing Point	-153°F	-71°F
Vapor Pressure at 77°F Storage Temperature	45 psia	14 psia

Materials compatibility constraints dictate aluminum propellant containers in lieu of the higher strength to weight stainless steels or titanium.

8.2 Space Storable Selection

Performance efficiency of FLOX/Methane has been demonstrated in isolated firings at design MR at 5.75 and low expansion ratio ($\varepsilon \sim 40$). When extended to larger ε the performance data given in Figure 14 are accepted as nominals. Three contracts funded by OART to Pratt and Whitney, TRW Systems, and Rocket Research, are now being undertaken to demonstrate performance capability of systems at low pressure, and to document realistic expected kinetic losses. Other phenomena such as cokeing and heat transfer are also being explored. In addition Lewis Labs is currently funding Marquardt Corp. to demonstrate extremely light weight composite carbon graphite combustion systems (Reference 7).

The common liquidous range CLR, of FLOX/Methane, although small, enables effective utilization of a simple, efficient and lightweight storage system. The extent of this CLR range is indicated in the pressure/temperature curves presented in Figure 15 (Reference 9). The freezing point of CH_4 is approximately 163°R, essentially independent of pressure. The boiling point of FLOX at 50 psi, approximately optimum pressure for effective CLR utilization, is 178°R. This ensures a workable common liquidous range of approximately 14°R, given thermal mixing.

FLOX/Methane liquidous range can be increased by increasing maximum allowable FLOX tank pressure. For example, a boiling point of 184°R is obtained at 70 psi vapor pressure, allowing a 20°F workable common liquidous range. Cost is approximately 1% gross weight above 50 psi vapor pressure storage. Alternately, the CLR can be extended by additives to methane which reduce the freezing point with only slight loss in performance (Reference 9). For example, a blend of 90%

methane and 10% ethane (Figure 16) has a freezing point of equivalent to a 20°R common liquidous range at 50 psi. The resulting I_{sp} degradation is only 1 1/2 secs. (A 55/45 mixture of methane/ethane lowers the freezing point to 133°R but results in a prohibitive 7 sec loss.) A 90/10 mixture at 70 psi further extends the CLR to 26°R.

Propellant heat sink capacity over the CLR determines surface staytime capability for a fixed performance insulation system. Specific heats of FLOX and methane are respectively .38 BTU/lb/°F and .81 BTU/lb/°F (Reference 10) which yields an effective propellant specific heat of .44 BTU/lb/°F (at 5.75 MR). The propellant heat sink capacity for propellant weights given in Table 2 are 1,420 BTU/°F and 2,380 BTU/°F, for the one and two man stages, respectively. Presumably, the entire CLR can be utilized in storage by subcooling the stage to the Methane freezing point. Total heat capacity for 14°R, 20°R and 26°R CLR are given below:

CLR (°R)	Heat Capacity 1 Man Stage (BTU)	Heat Capacity 2 Man Stage (BTU)	Comments	
			Methane/ Ethane Mixture (%)	Vapor Pressure (psi)
14°	20,000	33,400	100/0	50
20°	28,400	47,600	100/0	70
26°	36,900	61,900	90/10	70

From this table maximum "allowable" heat rates can be calculated for a fixed Mars surface staytime capability. Maximum heat rates for staytimes of 2 days, 1 week, 2 weeks, and 1 month are presented below. It is presumed that entry rates are increased by two orders of magnitude over nominal surface rates for a 1/2 hour period. 2 hours post entry heat input is presumed to be increased by one order of magnitude. In all cases a two day contingency is allowed. Design of insulation systems in section 9 are predicated on two week staytime allowances below.

		1 Man Stage			2 Man Stage		
Staytime	Equivalent Hours	14°F	20°F	26°F	14°F	20°F	26°F
2 days	186	108	153	198	180	256	332
1 week	306	65	93	121	109	156	202
2 weeks	454	44	63	81	74	105	136
1 month	868	23	33	42	38	55	71

MAXIMUM ALLOWABLE HEAT FLUX (BTU/HR.) FOR PASSIVE STORAGE AS A FUNCTION OF STAYTIME

9.0 SUBSYSTEMS DESIGN

Stage subsystems are considered on an item by item basis. Included are propulsion system, structures, equipment and instrumentation, and contingency allowances.

9.1 Propulsion Systems (Reference 8)

9.1.1 Engines and Related Subsystems

Engine thrust levels of interest range from 1,000 lbf to 8,000 lbf. Representative propulsion system data was available from Pratt and Whitney Aircraft (P & W) and Rocketdyne. Engine weight versus thrust obtained from P & W is given in Figure 17. P & W indicated that engine weight for Compound A/MHF-5 and FLOX/Methane are approximately the same. Design characteristics are 1,000 psia chamber pressure, 100:1 expansion ratio, 5.75:1 mixture ratio for FLOX/Methane, and 2.7:1 mixture ratio for Compound A/MHF-5. The overall dimensions of a 1,000 lbf thrust engine are about 16 in length and 9 in diameter. An 8000 lbf engine is approximately 61 in length and 31 in diameter. (These dimensions could be reduced by employment of extension bell or aerospike designs).

Main engine gimbal is utilized for pitch and yaw attitude control and small RCS thrusters for roll control. The estimated gimbal system weight for a 5,000 pound thrust engine is 10 pounds. The RCS hardware necessary for roll control is approximately 5 pounds. An additional 5 pounds of propellant is required. Weights at different thrust levels are estimated by pro-rating on a thrust basis.

Trapped propellant and residuals were calculated by assuming a feed line velocity of 10 fps. Representative trapped propellant and residuals weights given below for one man/2 stage designs.

	$FLOX/CH_4$		Comp. A/MHF-5	
	1st Stage	2nd Stage	1st Stage	2nd Stage
Trapped Propellant in Lines (lbs)	6.5	1.4	6.4	2.0
Residual Propellant in Tanks (lbs)	16.5	4.0	18.6	4.0
TOTAL (lbs)	23.0	5.4	25.0	6.0

9.1.2 Pressurization System (Reference 8)

Pressurization systems are sized independently for earth storable and space storable systems. Pressurization prior to entry is assumed to accommodate abort requirements. A single "false start" is allowed for ground pressurization contingency prior to ascent. In the two stage vehicle pressurization system, stage 1 was sized for a single start and stage 2 was sized for two starts, assuming a sufficiently long circular coast time to allow thermal equilibrium to occur. Pressurization was assumed to be maintained during coast. In the single stage design the pressurization system was sized for two starts and coast similar to that of the two stage vehicle second stage.

Helium, stored passively, is employed for earth storable pressurization. Container weight is sized by maximum 540°R (80°F) storage temperature. FLOX/CH_4 space storable pressurant is Tridyne O_2 + H_2 + He catalytically ignited, stored in the insulated container at 170°R. Pressurization includes 5 psia for net positive suction head (NPSH) (to prevent pump cavitation), 4 psia for line losses, and 3 psia for acceleration losses.

9.2 Structures

9.2.1 Propellant Tanks

Propellant container sizing factors are summarized in Table 5 and 6. In calculating propellant volumes, 5% ullage was assumed for earth storables and 10% ullage for space storables to accommodate variable densities over the CLR temperature range.

Proof pressure test is 1.10 x [Maximum relief valve pressure + hydrostatic head]. An additional safety factor of 1.10 x [proof pressure] is required for design to material yield stress. Design yield stress for aluminum is chosen as 55,000 psi with .010 in minimum gage.

Maximum tank operating pressure is a function of several factors including storage vapor pressure, operating vapor pressure, NPSH and acceleration and line losses. Based on the desirability of maintaining fuel vapor pressure at a minimum of 14.7 psia during prelaunch and launch storage, (section 5.1) the selected storage temperature is the maximum achievable without exceeding maximum oxidizer operating pressure or minimum propellant container gage pressure, whichever is smaller. Maximum storage vapor pressure is equal to maximum operation vapor pressure plus operational pressure penalties (NPSH, acceleration and line losses), not additive to vapor pressure in the storage case.

Ideally a storage temperature is chosen to satisfy both conditions simultaneously. Otherwise, a temperature is chosen to match the "best" combination of storage pressures.

For compound A/MHF-5 the optimum storage temperature is determined from the vapor pressure/temperature charts (Figure 18) to be 77°F. Design temperatures are 75°F and 79°F for fuel tank compression and oxidizer tank tension, respectively.

For FLOX/Methane a maximum storage temperature of 187°R is selected which yields a FLOX vapor pressure of 67 psia corresponding to maximum operating pressure at 178°R (50 psia vapor pressure, plus NPSH, acceleration and line losses).

The results of these design approaches are given in Table 6 for both classes of vehicles. Note that in the Compound A/MHF-5 stages the maximum duty cycle pressure in all cases conforms to the ideal pressure margins with only negligible differences. The situation in the FLOX/Methane propellants is slightly less favorable. Here the Methane tank goes into 8 psia compression and in the larger tanks, minimum gage oxidizer pressure is exceeded. A 20% factor for added stiffeners to maintain compression is assumed for methane fuel tanks.

Tank weights are calculated assuming a 20% penalty for weld lands and attachments. The tank design pressures, weights, and volumes are summarized in Table 6.

9.2.2 Insulation

Earth Storables - A potential problem of earth storable propellants is MHF-5 freezing, especially local icing. For the purposes of estimating insulation weight, it is presumed that the spacecraft is at the mean temperature of the surface environment (-45°F) and that passive insulation on the tank and interstage structure is to maintain propellants within temperature excursions of ±15°F. It is assumed that 50% of the heat input is from supports and plumbing and 50% is from insulation. Insulation material (non evacuated) is aluminum coated polyeurethane foam with a density of 3 lb/ft^3 and a thermal conductivity equal to .007 BTU in/hr ft^2°F at 15 psi. Calculated insulation weights are given in Table 7, and are so low as to be almost insignificant.

Space Storables - The insulation container is sized for a nominal staytime of two weeks, corresponding to design heating rates of 44 BTU's/hr and 74 BTU's/hr for 1 and 2 man stages, respectively (obtained in Section 8.2). Half of the heat input is assumed to pass through heat shorts, and half through the insulation. Based on respective 1 and 2 man vehicle areas, the allowable heating rates for both vehicles are given in Table 8. Assuming the average temperature of the landed vehicle to be 20°R above the surface mean of 415°R, the mean temperature difference is 260°R. The required design thermal coefficient is, therefore, $\sim 5 \times 10^{-4}$ BTU/ft^2/hr/°F. At 10 mb Mars surface pressure this performance is achievable with evacuated polyuerethane foam spaced aluminum coated polyester radiation shielding. Insulation characteristics of this material at 15 psi and vacuum are given in Table 9 (Reference 11). Performance degradation under atmosphere pressure is shown in Figure 20 from which the 10 mb coefficient is extrapolated.

Conductivity at 10 mb is estimated to be 4×10^{-4} BTU-in/hr ft^2°F. Assuming a factor of two degradation from edge losses, a 2 inch layer of insulation must be employed. Density is 1.6 lbs/ft^3 for insulation and .03 lbs/ft^2 for the vacuum barrier. Based on these properties, insulation penalties accrued on the descent stage are given in Table 8. Insulation at the manned capsule/forward stage interface cannot be discarded at launch, and hence is charged as an ascent weight penalty.

As shown in Figures 10 and 11 principal heat leak sources are stage 1 base supports, manned capsule/forward stage structural ties, and instrument unit/forward stage connections. The ascent vehicle is supported along stage 1 tank extensions by circumferential support cones tapered to points at the insulation interface. The supports are designed for a maximum loading of 10 g's and employ glass fiber tension ties (200,000 psi yield) through the insulation, with a thermal conductivity of 7 BTU-in/hr ft^2°F (Reference 11). Designing to 100,000 psi, the total support cross section (through the insulation) is 1/2 in^2. Conservatively assuming 2 in. effective support length and a temperature differential of 260°F across the supports, the heat input is 6 BTU/hr.

Manned capsule/forward stage support penetrations through the insulation are a minimum gage glass fiber thrust cone, and 2 ascent capsule supports. Redundant capsule descent supports are retracted after touchdown. For 4,300 lbf first stage thrust, and 1,500 lbf second stage thrust, thrust cone and support sections are minimum gage, approximately 1/2 square inch. As in the previous case, heat leaks can be limited to 6 BTU/hr.

Power, telemetry, and wiring connections are disconnected by plug removal during surface staytime. Alternately transformer gaps are employed to eliminate physical connections entirely. 12 BTU's/hr is maximum allowable heat leak rate for auxillary penetrations for storage monitoring.

Summing heat inputs to the one man stage:

	BTU's/HR
Insulation	22
Supports	6
Structural Capsule/ Stage Connections	6
Capsule/Stage Instrument Systems Interfaces	12
	44

Similar calculations for the 2 man stage are obviously conservative.

9.2.3 Interstage, Thrust Structure, and Base Heat Protection

Interstage loads are estimated for the worst of three conditions:

1. 10 g entry loading/manned capsule supported
2. 3 g abort loading/manned capsule unsupported
3. 1 g ascent/thrust loading

Stage 1 engine thrust loads are transmitted directly through the tank skins minimizing unpressurized structure except between stages, and at the stage capsule interface. The thrust structures are stiffener reinforced and are mounted directly to the tankage. Base heat protection, where required, is provided by by ablator coated honeycomb panels for which combined weight of 2 lb/ft^2 is assigned.

9.3 Equipment and Instrumentation

Based on previous stage designs a 10% dry weight contingency for equipment and instrumentation is a conservative estimate, particularly in view of major technological advances such as integrated circuitry anticipated in the next several years.

9.4 Contingencies

20% of MEM stage dry weight is allocated for small systems backup and bypass components.

10.0 SPACE STORABLE REFRIGERATOR SYSTEM DESIGN

Missions of greater than two week duration must employ active surface refrigeration. A refrigerator system carried on the descent vehicle can be utilized during the entire mission, or especially provided for operation during the post landing phase. This choice is significant as refrigeration system selection and performance are strongly dependent on operational lifetime requirements. Conservatively assuming an extended lifetime capability greater than 500 hours, it is estimated in Reference 12 that for 175°R(~100°K)FLOX/Methane storage temperature, 12-13 watts of refrigeration (35-38 BTU's/hr) can be supplied for about 440 watts electric. (The power/temperature performance relationship is shown in Figure 20.) With the insulation systems designed for two week staytime passive storage, a 12 watt (average) refrigeration system can extend the one man stage staytime indefinitely and two man stage staytime to over 1 month. Increasing the CLR to 26°F from 14°F could effectively double these staytimes. For example a factor of two improvement in insulation coupled with a 20°F CLR pressure vessel design increases 1 man-stage staytime to 1 month.

The descent stage penalty for an active refrigeration system (excluding insulation) is less than 250 lbs. Assuming 85% conversion efficiency, approximately 500 watts constant power is required, or a 1 kw daytime cycle if solar cells are employed. Using solar cells, performance on the Martian surface is rated at 100 lbs/kw. Allowing a 50% contingency for fixed viewing angle degradation, solar cell weight is 150 lbs. Refrigerator weight for a 24 watt system consistent with the 1 kw average power unit is approximately 80 lbs. With 20 lbs (10%) added contingency, total weight is 250 lbs.

11.0 MSSR/MEM COMMONALITY

The MEM propulsion system can be utilized in association with an unmanned Mars surface sample return (MSSR) payload in lieu of a manned capsule. The objectives of such a mission are twofold:

1. Scientific - to discover lifeforms and determine pathological characteristics prior to manned landing, and

2. MEM Qualification - to achieve MEM reliability by extensive unmanned operation via MSSR type missions.

Table 10 shows velocities achieved by unmodified two stage earth storable and single stage space storable vehicles for fixed 42 lb payload derived in Reference 2. All vehicles achieve velocities greater than 31,000 fps sufficient for low energy classes of dual and triple planet flyby missions. Higher energy (or payload) missions in the 36,000 fps to 40,000 fps class require an added stage. Table 11 shows MEM launch capability with the added stage plus payload gross weight equal to the weight of the manned capsule. Payloads range from 77 to 212 lbs for 36,000 fps velocity and 27 to 102 lbs for 40,000 fps velocity. Performance characteristics of the added stage estimated from Table 1 are summarized below:

	Earth Storable		Space Storable	
Gross Weight (Including Payload)	I_{sp} (secs)	λ	I_{sp} (secs)	λ
700 lbs	344	.883	400	.874
1,300 lbs	348	.887	406	.869

12.0 RECOMMENDATIONS FOR FURTHER STUDY

Prospective requirements for further technological research and development have in the course of this study been identified in the following areas:

- Propulsion systems - high performance 1-10 klbf thrust earth storable and space storable engines.
- Propellant handling - earth storable and space storable propellant packaging and storage, effectively utilizing propellant CLR.
- Refrigeration systems - small, reliable, light weight refrigeration systems operating at about 100°K within narrow (i.e., ± 1°K) temperature deadband cycles.
- Insulation and thermal isolation design - design of insulation jackets, and structural support and subsystems interfaces consistent with non-vented storage requirements.
- Stage design - development of common bulkhead design concept for CLR storage.
- Advanced materials - investigation of applications for light weight composite materials for unpressurized skirt and shroud areas.

13.0 ACKNOWLEDGEMENTS

The author should like to acknowledge and thank Messrs. C. Bendersky, R. Gorman, E. D. Marion, A. E. Marks, and J. J. Schoch for the invaluable consultation provided during the course of this study. Mr. Bendersky made extensive recommendations and inputs in the areas of propellant selection and propulsion systems. Messrs. Gorman and Marion were consulted in areas of refrigeration, thermal control and insulation systems. Mr. Marks performed extensive supporting analysis in propulsion systems selection and design. Section 9.1 is essentially a summation of a letter by Mr. Marks to the author which contained data on engine weight, attitude control, propellant residuals, and pressurization systems. Mr. Schoch did extensive trajectory analysis and tradeoff studies for initial thrust to weight optimization and ΔV estimation.

1013-MHS-llf

M. H. Skeer

Attachments

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4. Eilertson, W. H., Personal Communication with the author.

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12. Gorman, R., and Marion, E. D., "Trip Report - Environmental Control and Life Support System Discussions with McDonnell-Douglas and Garrett Air Research," Bellcomm Memorandum for File, December 29, 1967.

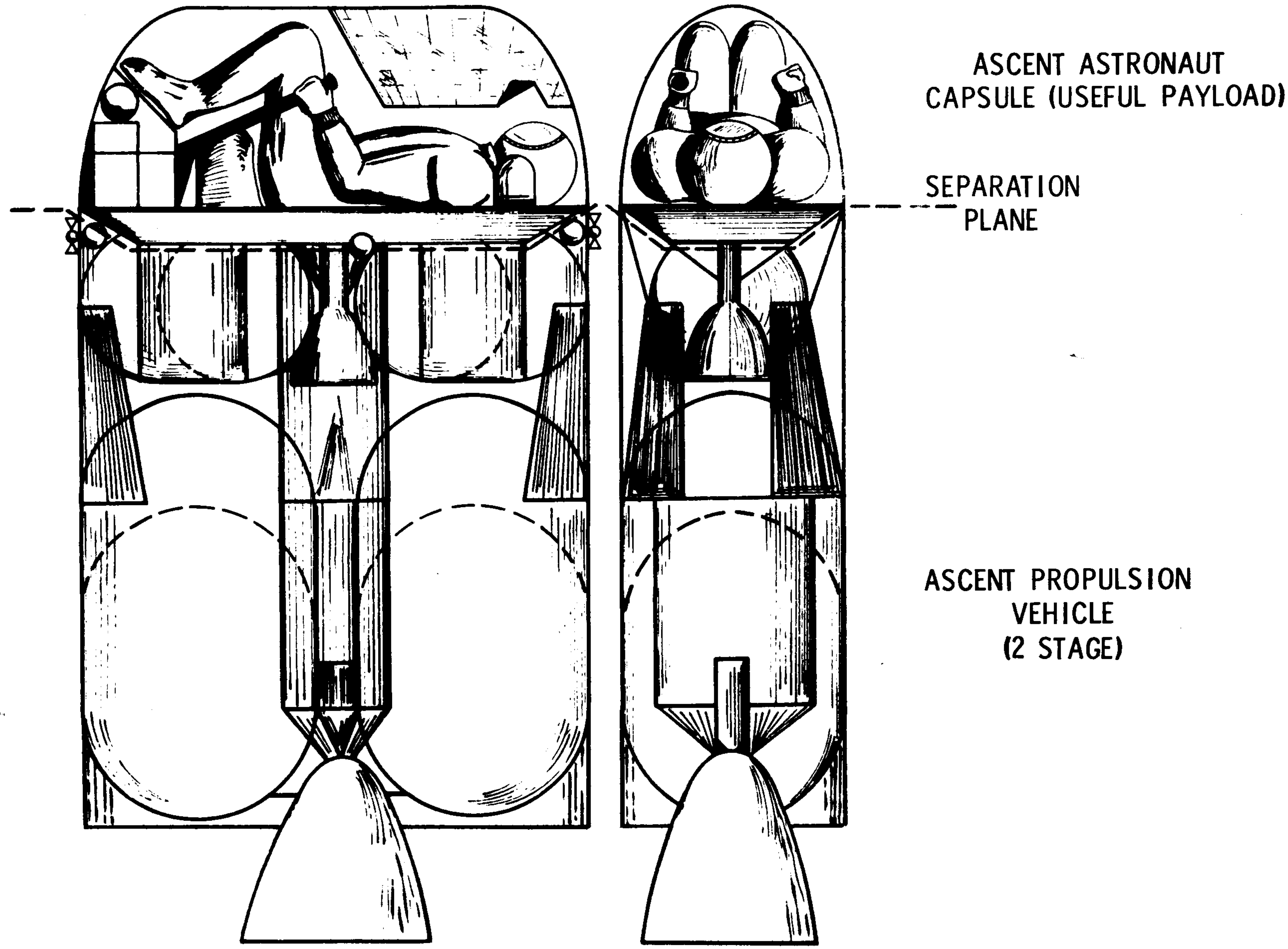
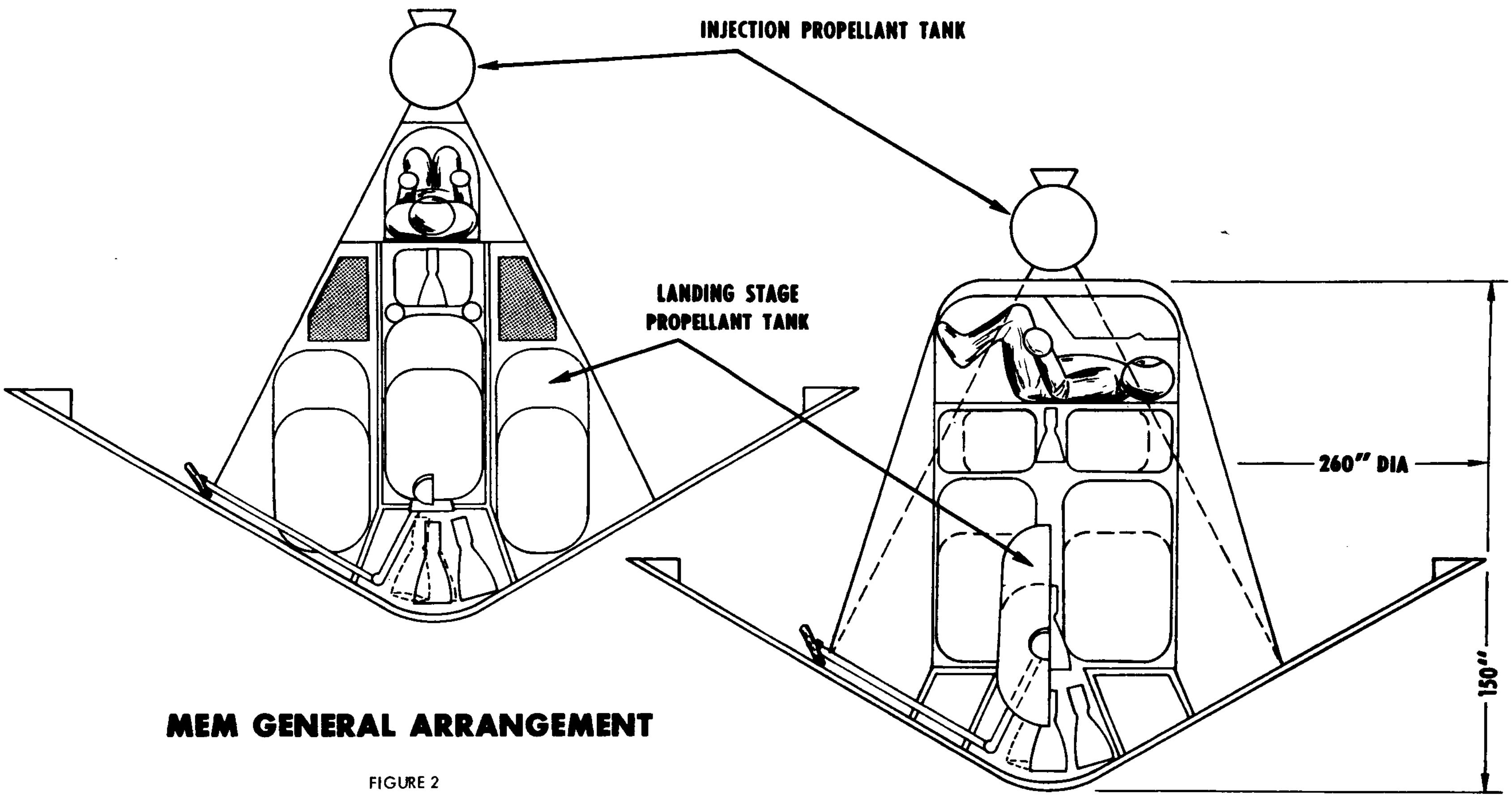


FIGURE 1 - MEM/ASCENT STAGE



MEM GENERAL ARRANGEMENT

FIGURE 2

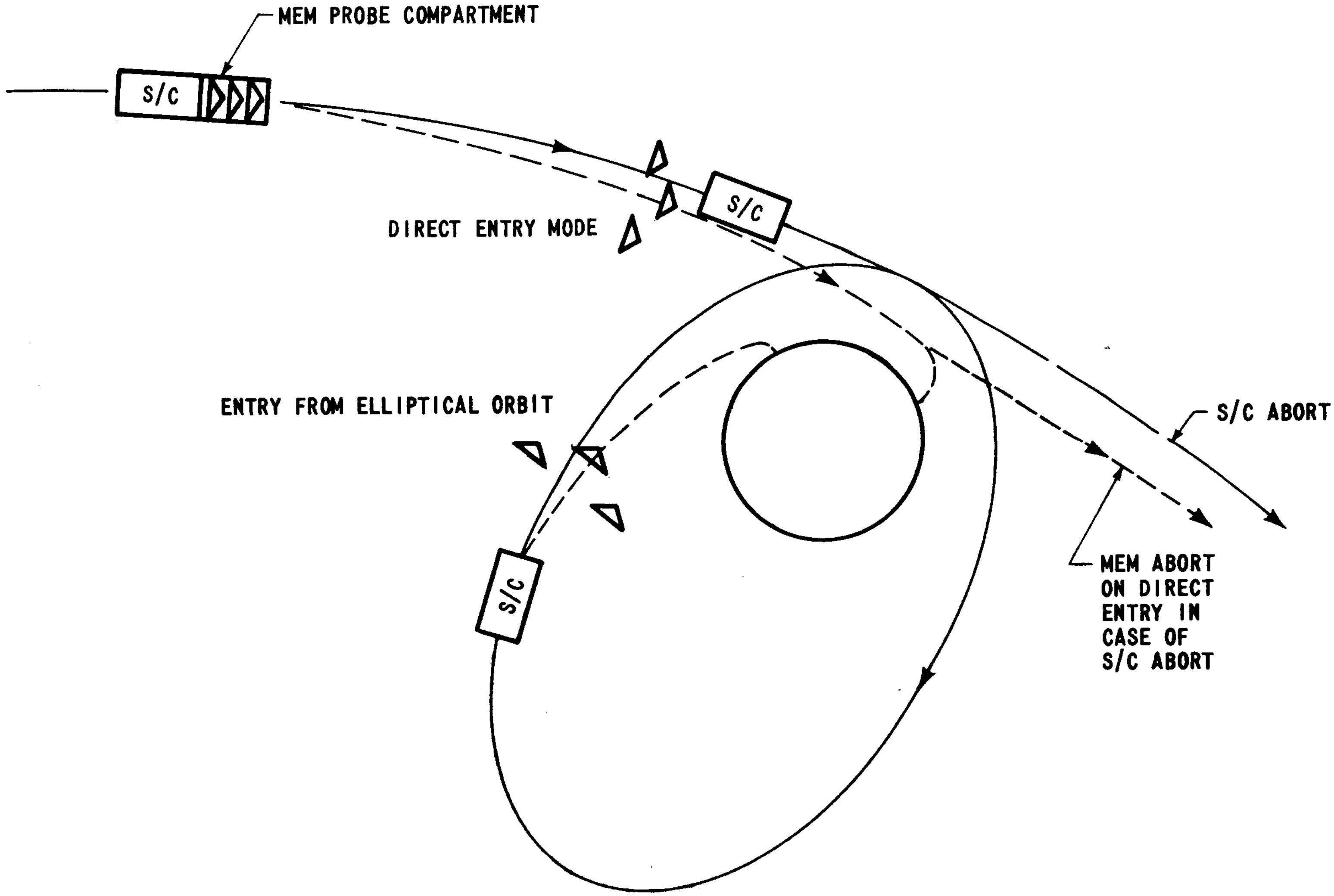


FIGURE 3 - DIRECT vs. ELLIPTICAL ORBIT MEM ENTRY MODE

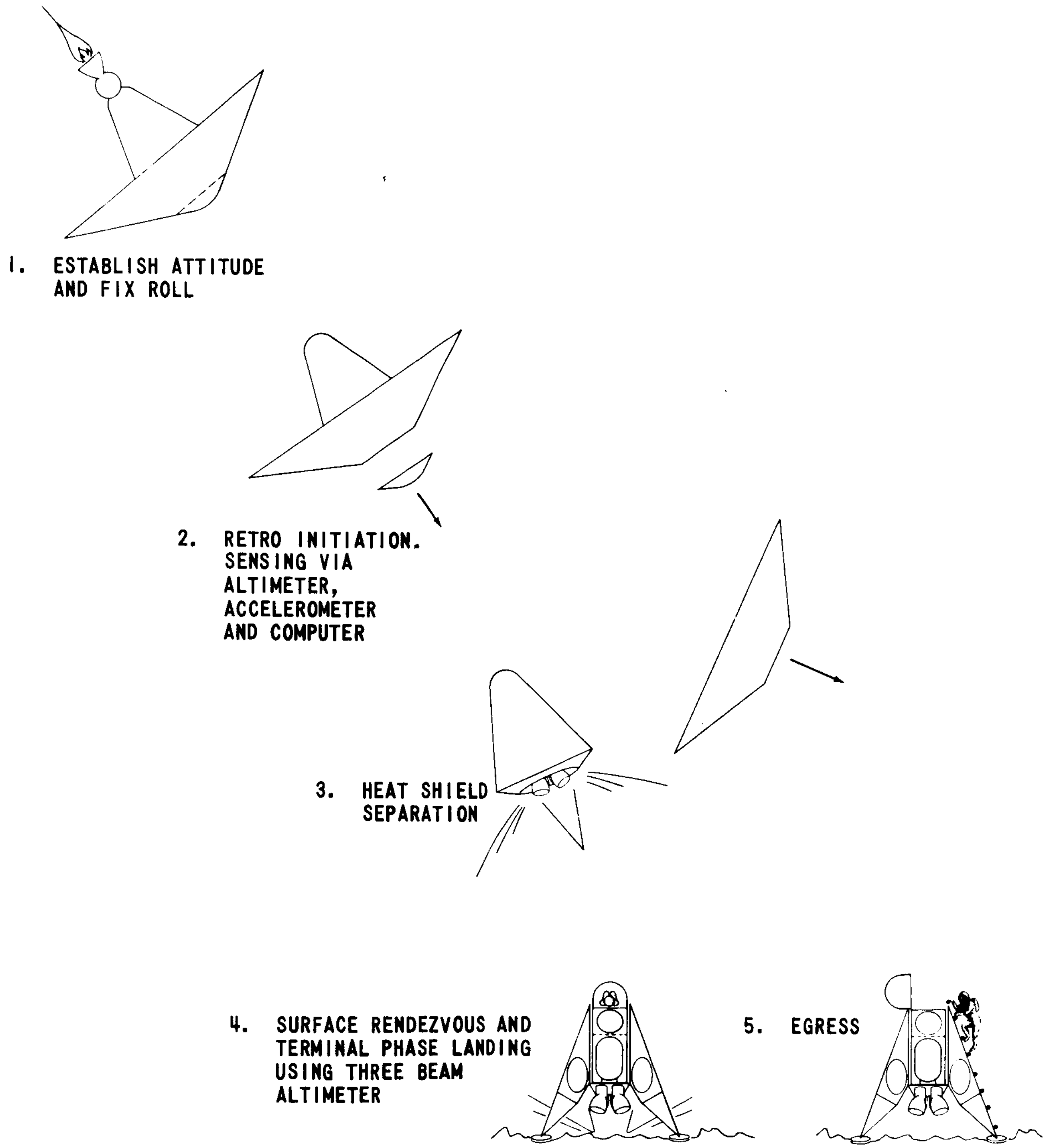


FIGURE 4 - MEM ENTRY SEQUENCE

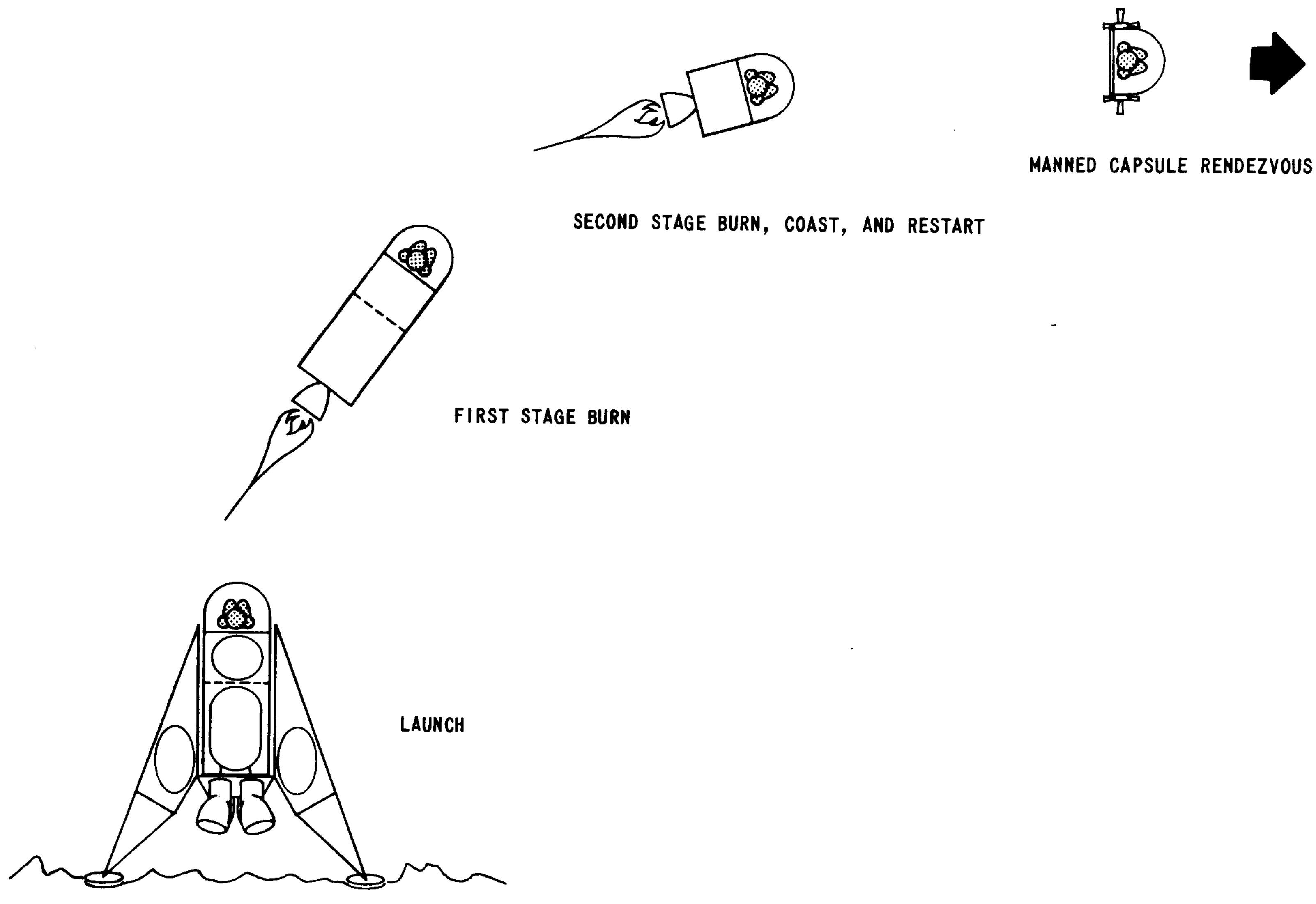


FIGURE 5 - ASCENT SEQUENCE FROM SURFACE OF MARS

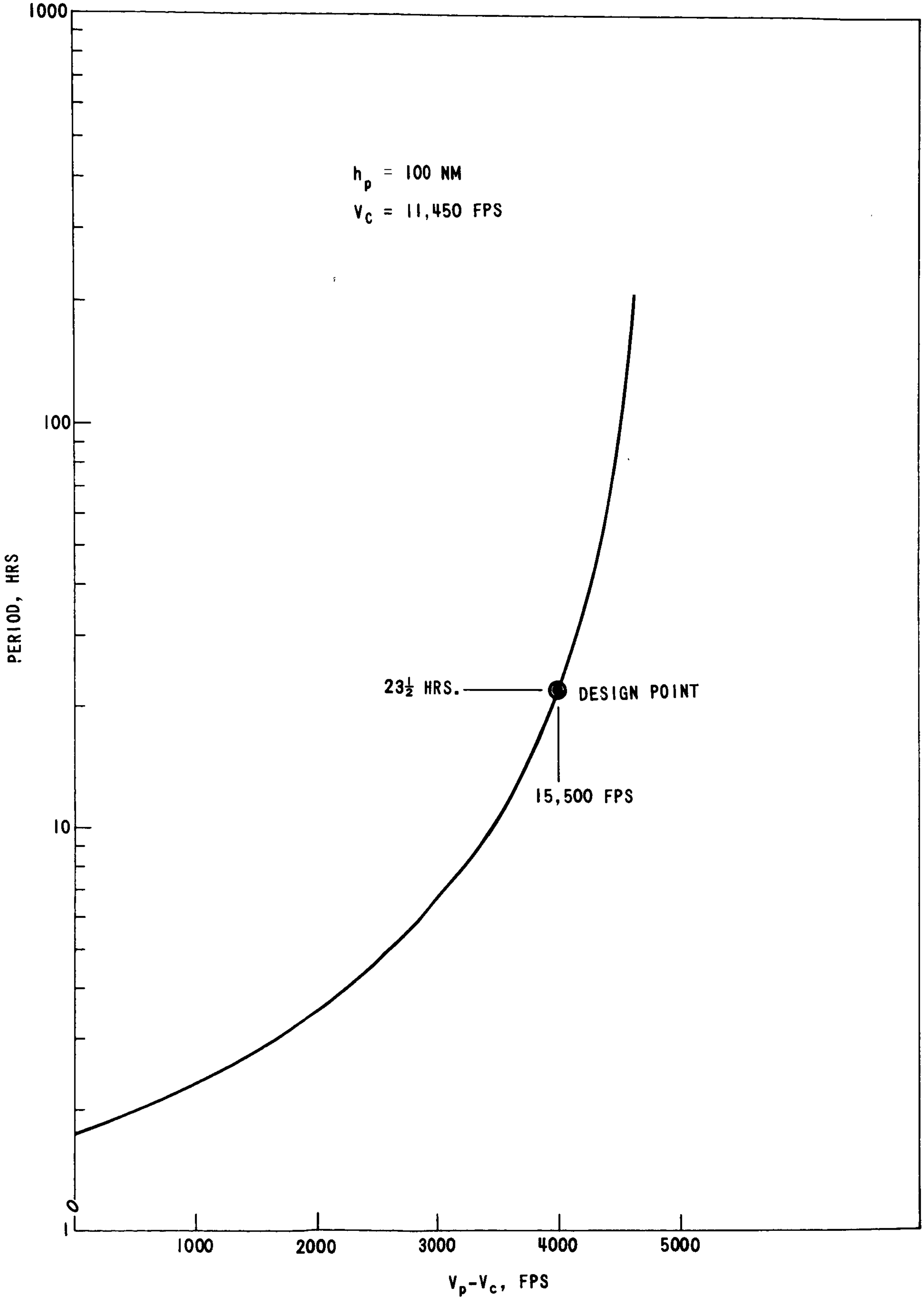


FIGURE 6 - ELLIPTICAL PARKING ORBIT AT MARS

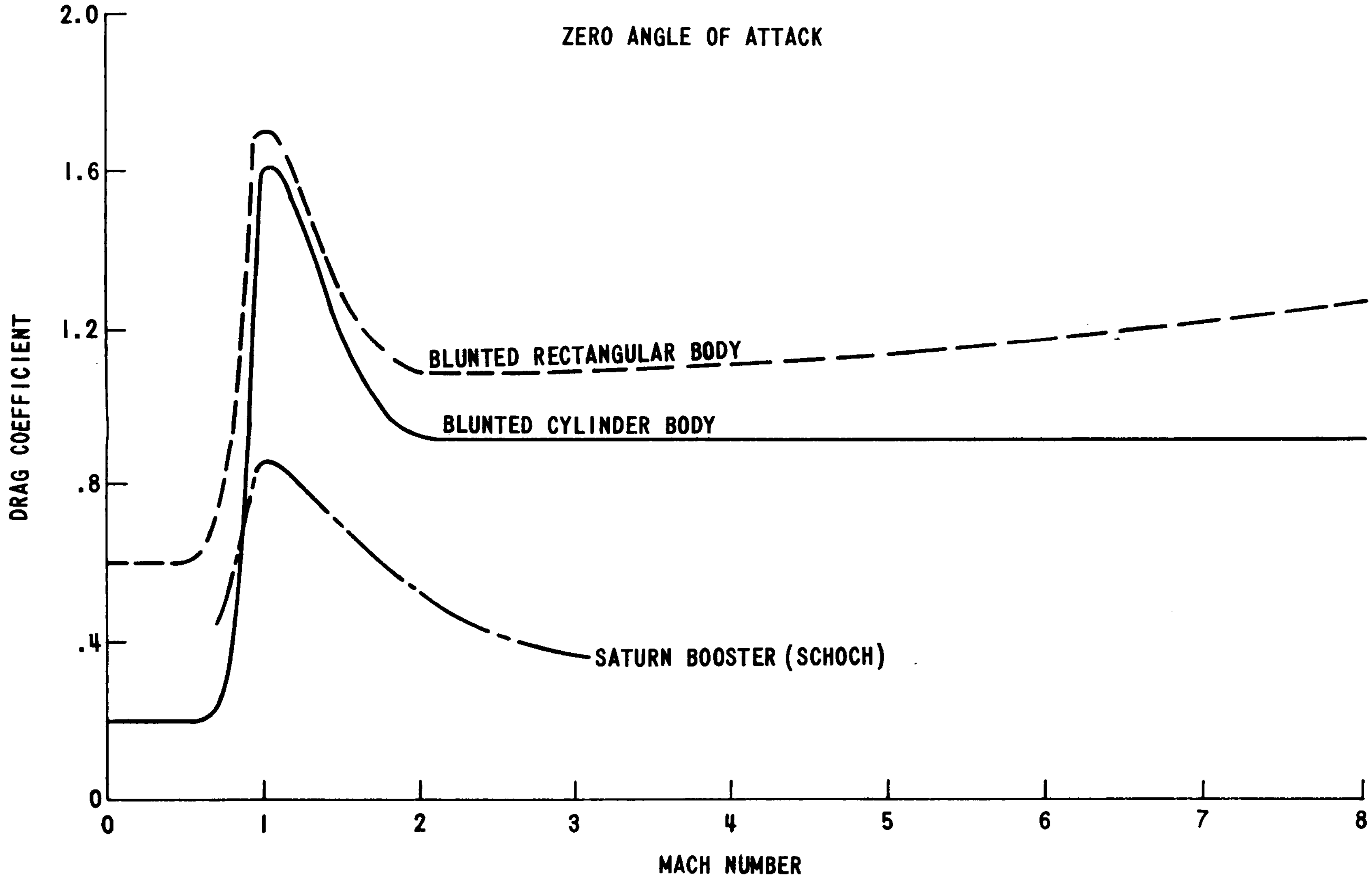


FIGURE 7 - ESTIMATED DRAG COEFFICIENT FOR BLUNTED RECTANGULAR AND CYLINDRYCAL BODIES

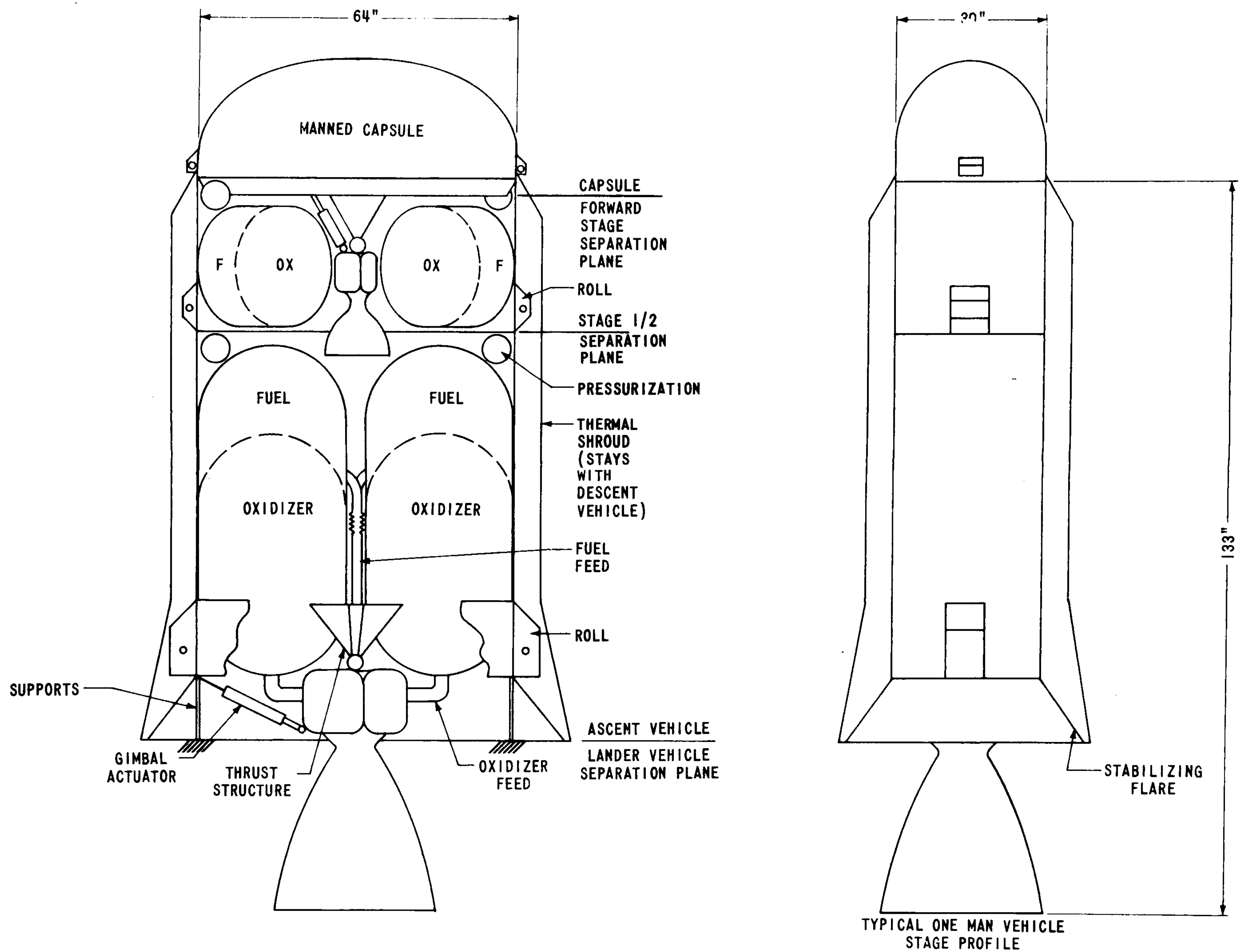


FIGURE 8 - ONE MAN/EARTH STORABLE MARS ASCENT VEHICLE

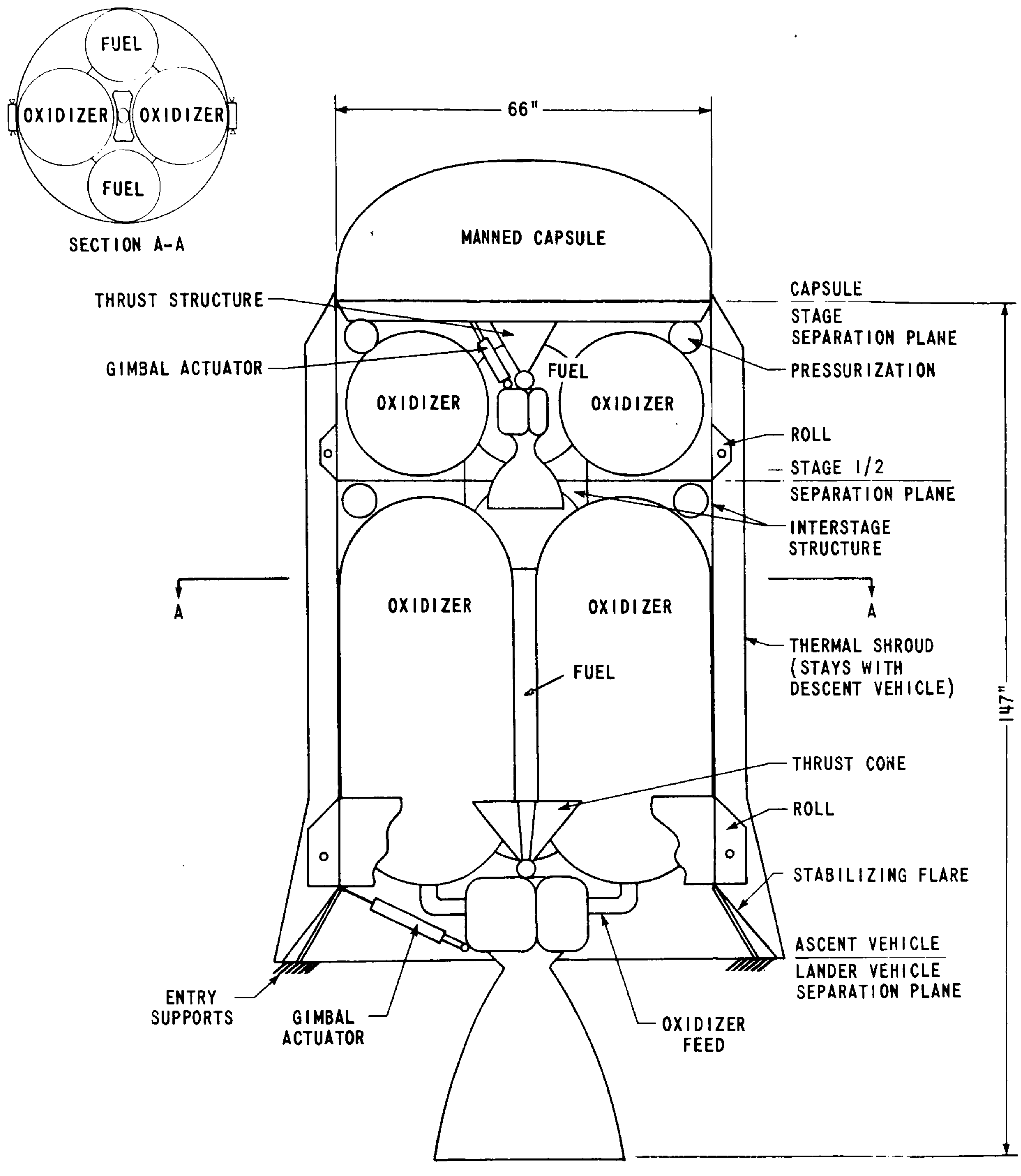


FIGURE 9 - TWO MAN/EARTH STORABLE MARS ASCENT VEHICLE

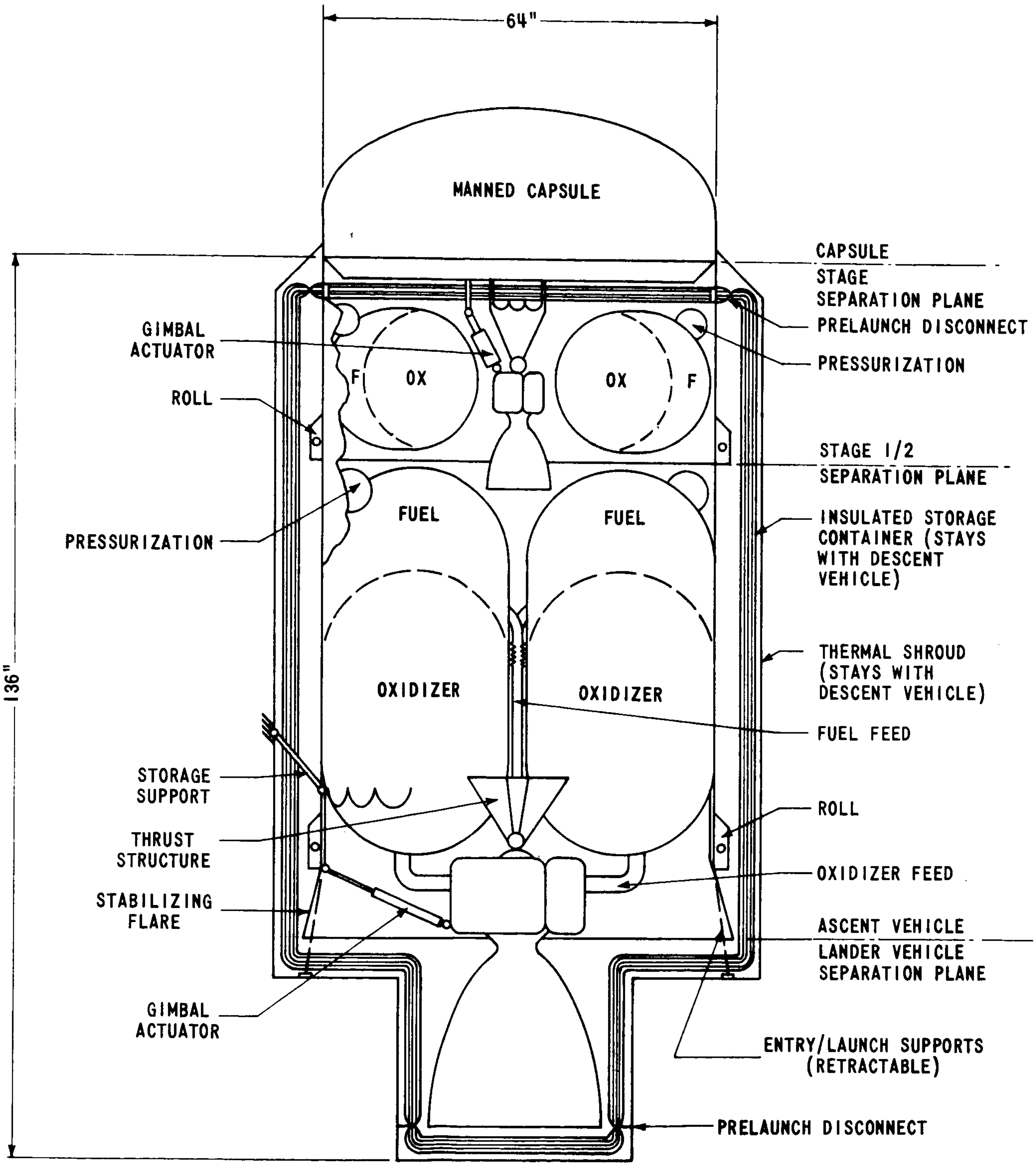


FIGURE 10 - ONE MAN/SPACE STORABLE MARS ASCENT VEHICLE/TWO STAGE DESIGN

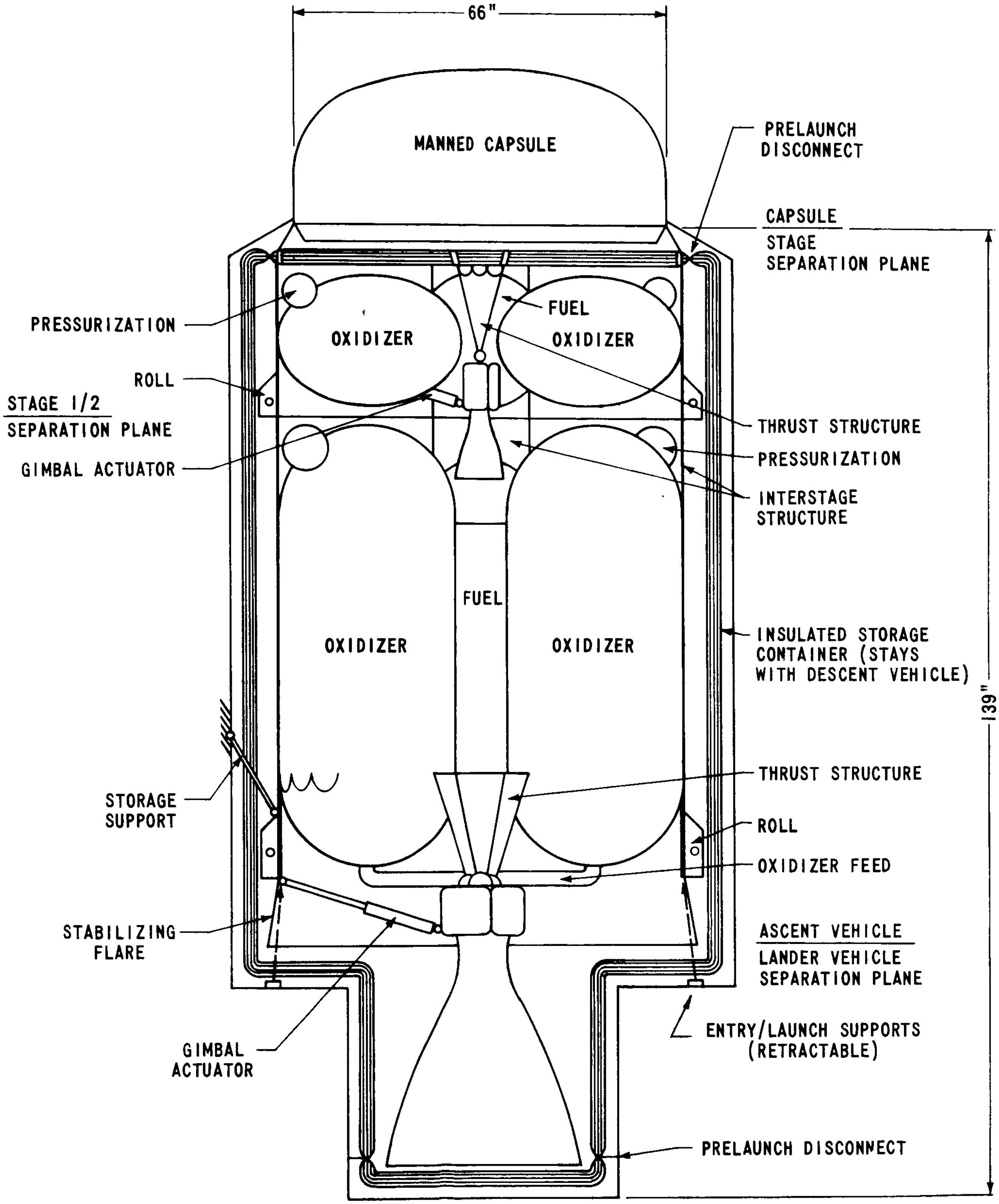


FIGURE 11 - TWO MAN/SPACE STORABLE MARS ASCENT VEHICLE/TWO STAGE DESIGN

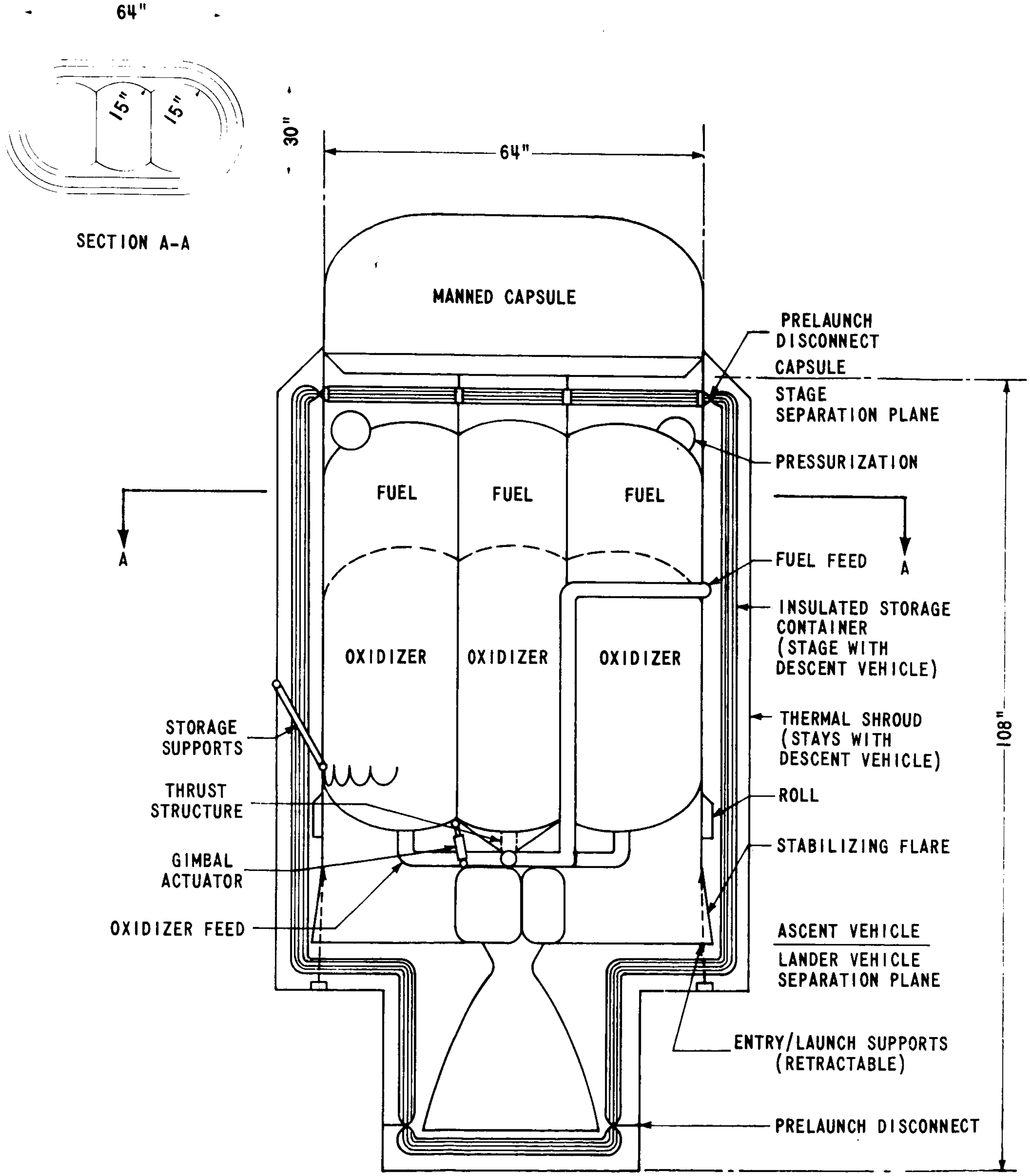


FIGURE 12 - ONE MAN/SPACE STORABLE MARS ASCENT VEHICLE/SINGLE STAGE DESIGN

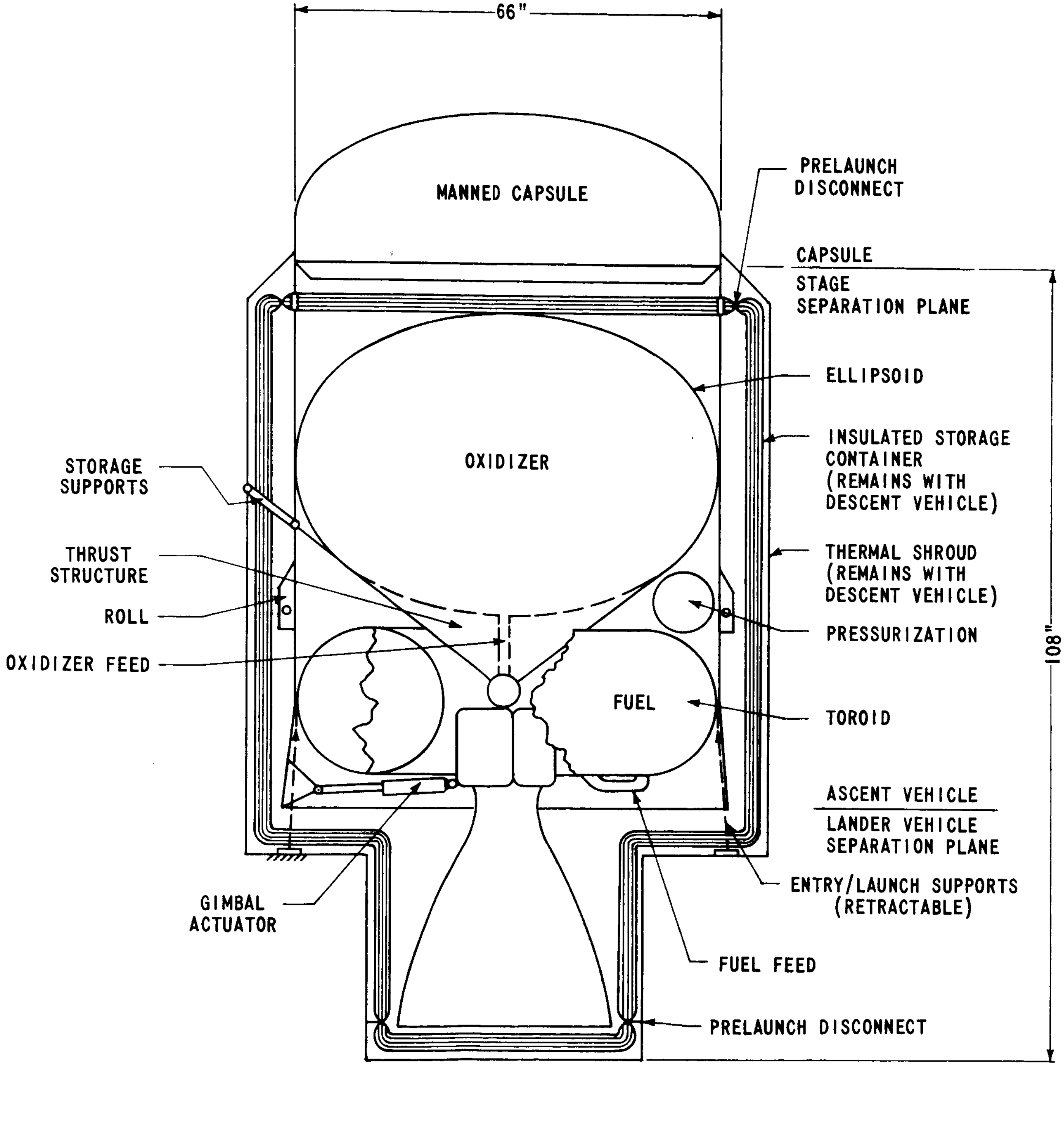


FIGURE 13 - TWO MAN/SPACE STORABLE MARS ASCENT VEHICLE/SINGLE STAGE DESIGN

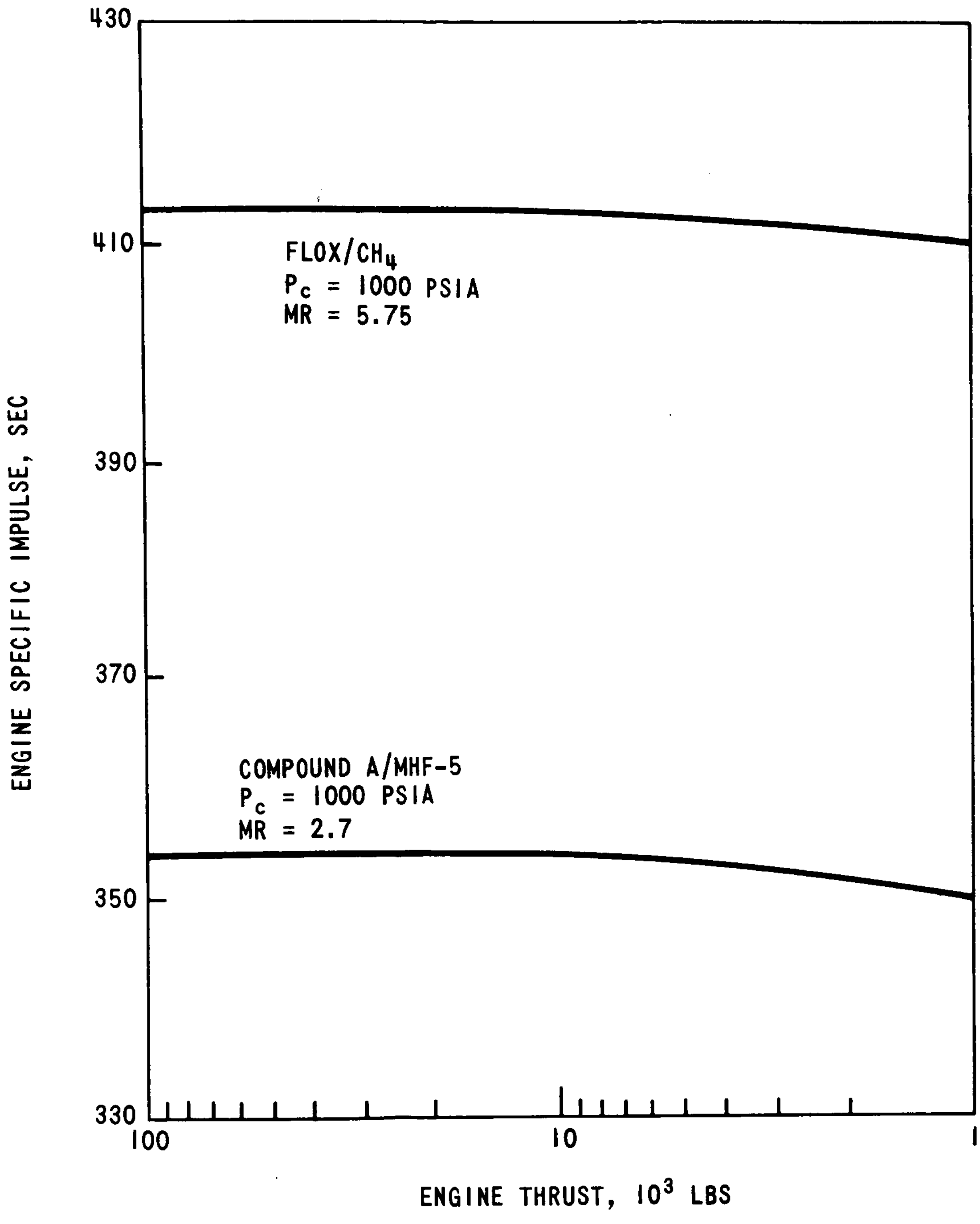


FIGURE 14 - ESTIMATED PROPELLANT PERFORMANCE AS A FUNCTION OF THRUST

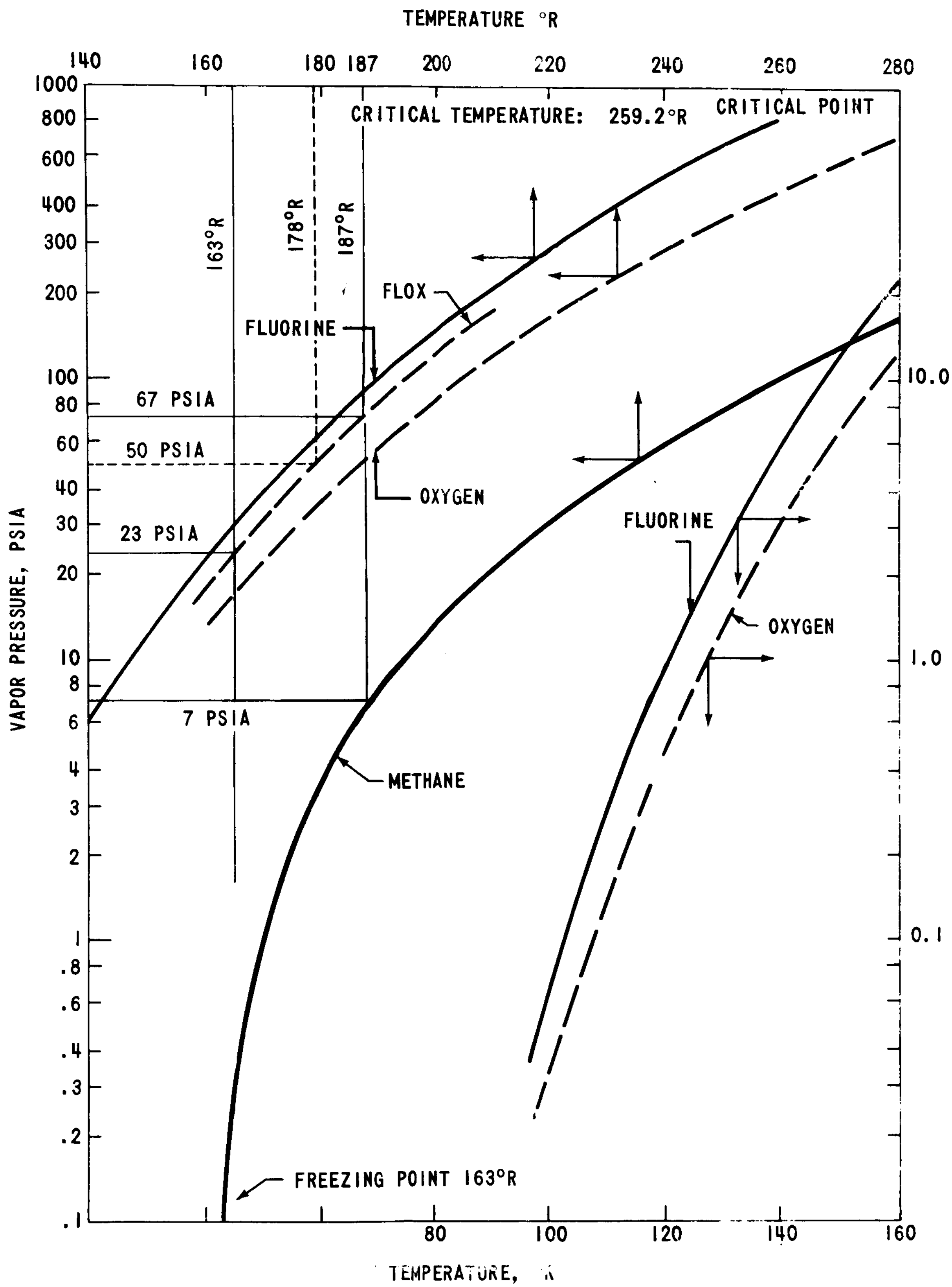
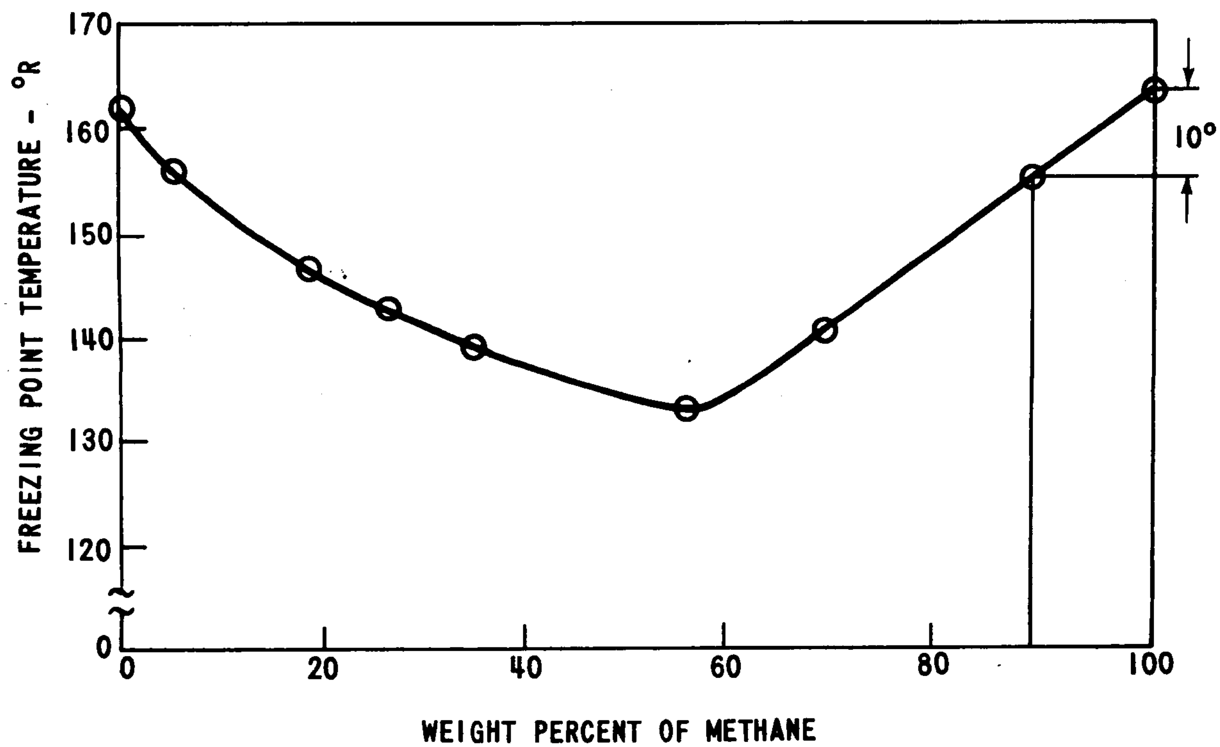


FIGURE 15 - VAPOR PRESSURE OF LIQUID METHANE FLUORINE AND OXYGEN



NOTE: THE HEAT CAPACITY OF THE FUEL IS TAKEN BETWEEN THE FREEZING POINT AND THE 150 PSIA BOILING POINT

FIGURE 16 - EFFECT OF BLENDING ON THE FREEZING POINT AND COOLING CAPACITY OF HYDROCARBON FUELS (METHANE-ETHANE)

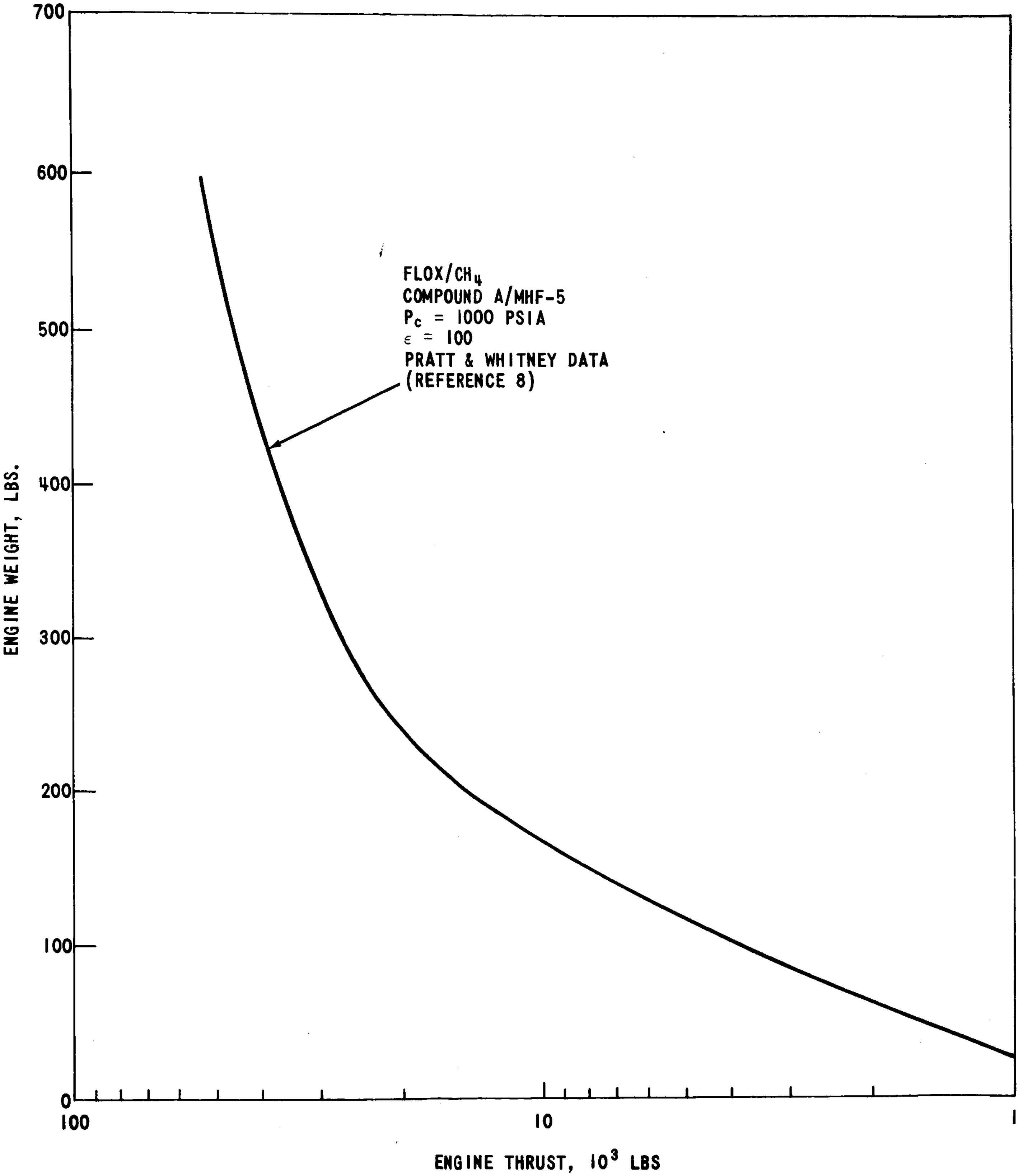


FIGURE 17 - ESTIMATED ENGINE WEIGHT AS A FUNCTION OF THRUST

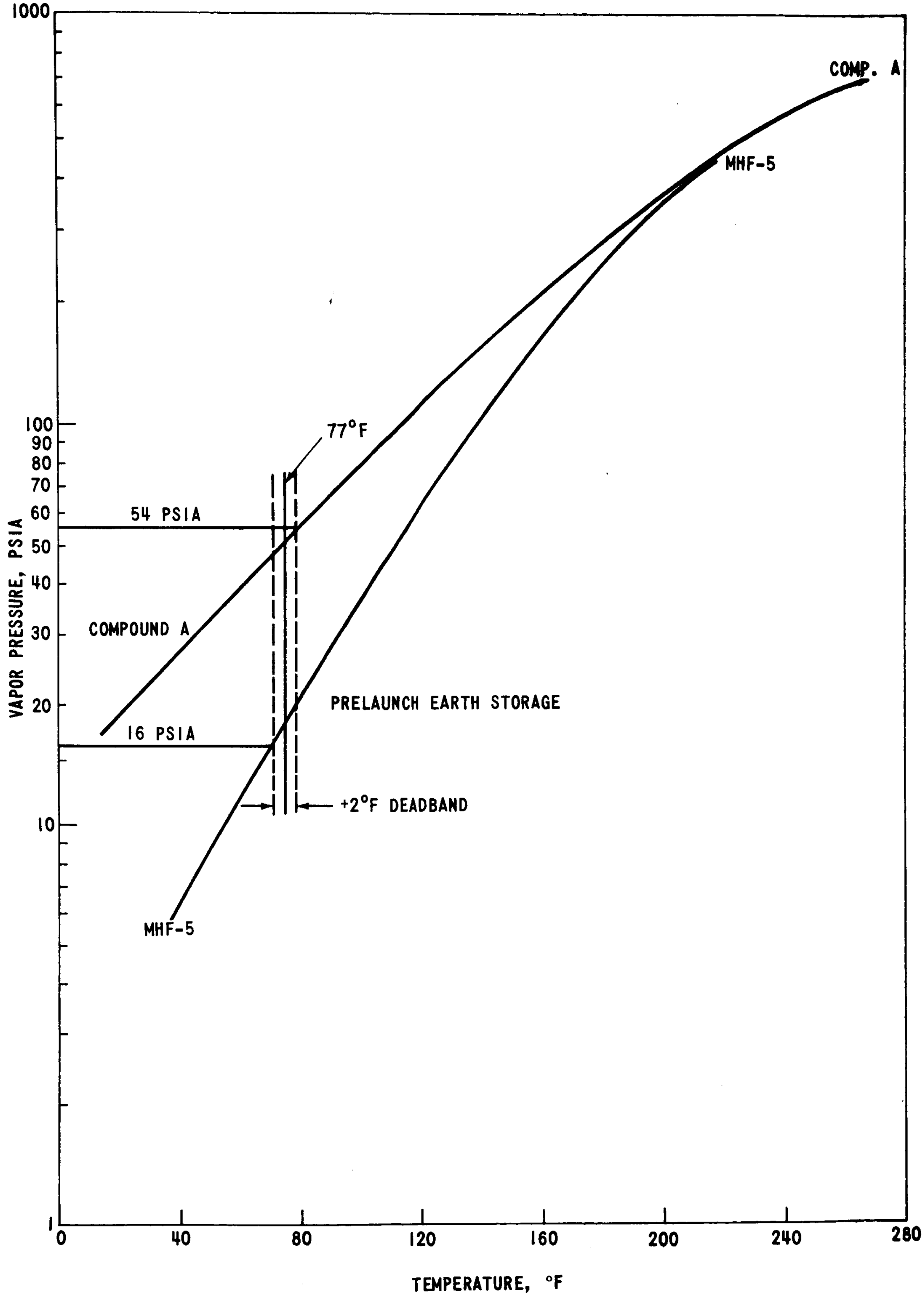


FIGURE 18 - EARTH STORABLE VAPOR PRESSURE TRADEOFF

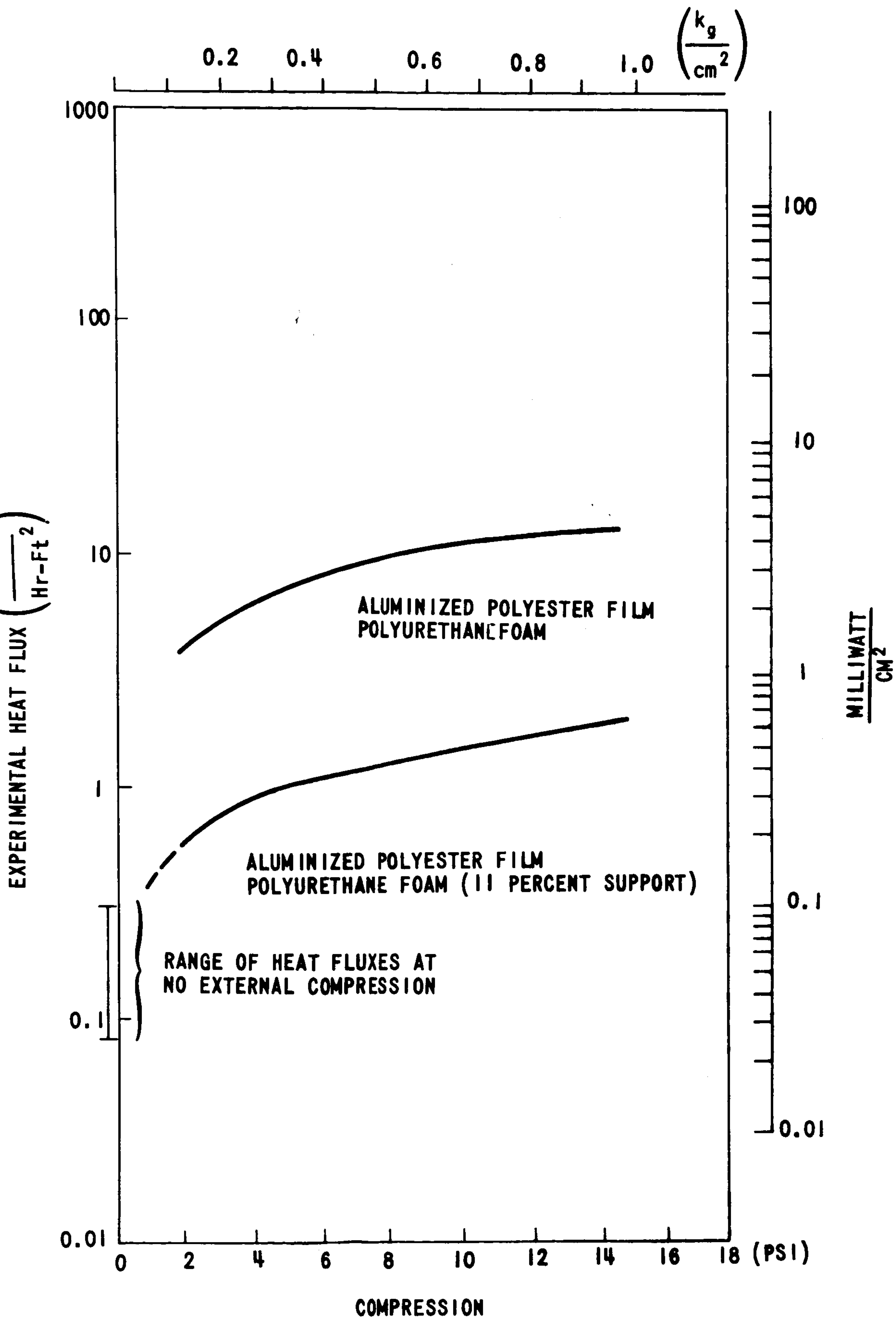


FIGURE 19 - EFFECT OF MECHANICAL LOADING ON THE HEAT FLUX THROUGH MULTILAYER INSULATIONS

		REFR. WATTS			REFR. WATTS
1.	ADL CRYODYNE	0.6	8.	AIR PRODUCTS-JT	2
2.	AIR PRODUCTS-MIT (CASCADE)	0.75	9.	CRYOGENERATOR	875
3.	ADL CRYOSTAT	75	10.	CRYOGENERATOR	3500
4.	ADL CRYOSTAT	200	11.	AIR PRODUCTS	0.25
5.	CRYOVAC	700	13.	AIR PRODUCT-JT	0.35
6.	NORELCO	10	14.	SBRC	5
7.	MALAKER	5	19.	HUGHES	15

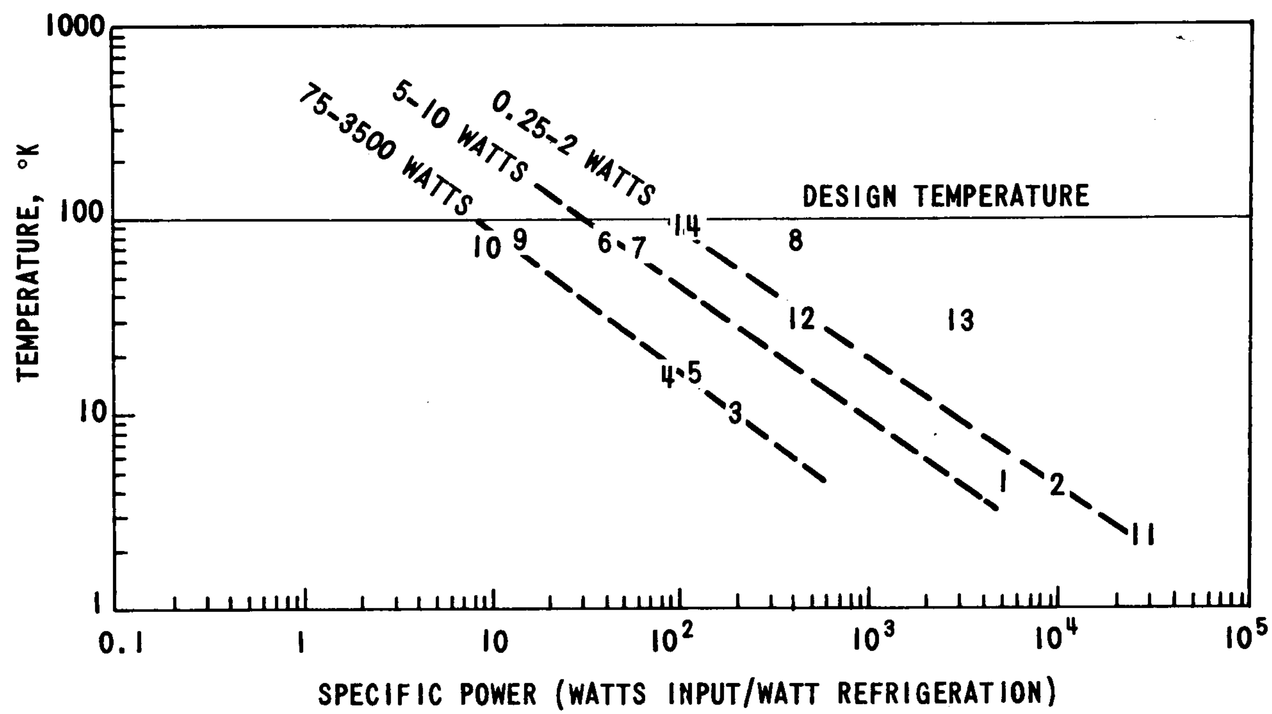


FIGURE 20 - SUMMARY OF TEMPERATURE vs. SPECIFIC POWER OF PRESENTLY AVAILABLE CRYOGENIC REFIGERATORS

TABLE I SUMMARY OF VEHICLE CHARACTERISTICS

VEHICLE	ONE MAN ASCENT (700 LBS)					TWO MAN ASCENT (1300 LBS)				
PROPELLANT	COMPOUND A/MHF-5		FLOX/METHANE			COMPOUND A/MHF-5		FLOX/METHANE		
NUMBER OF STAGES	2		2		1	2		2		1
STAGE	1	2	1	2	1	1	2	1	2	1
INITIAL T/M	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕	1 g ⊕
Isp (ϵ = 100)	350	344	405	400	406	353	348	410	406	412
λ' (FIRST ITERATION)	.919	.883	.907	.861	.920	.929	.889	.916	.874	.925
λ_o (INITIAL ASSUMPTION)	.920	.880	.900	.870	.910	.930	.900	.910	.880	.920
MIXTURE RATIO P_c = 1000 PSI, PUMP FED AND/OR TRANSPIRATION COOLED	2.5	2.5	5.75	5.0	5.75	2.5	2.5	5.75	5.0	5.75
THRUST (LBF)	5000	1500	4300	1500	4450	9000	2700	7300	2600	7600
ENGINE T/W	49	44	47	44	48	54	45	52	45	52
STAGE VELOCITY (FPS) ($\Delta V_1 + \Delta V_2$ = 19,000 FPS)	11,600	7,400	11,400	7,600	19,000	11,900	7,100	11,600	7,400	19,000
STAGE GROWTH FACTOR	3.35	2.23	2.81	2.08	5.95	3.20	2.13	2.82	2.00	5.70
TOTAL GROWTH FACTOR	7.45		5.85		5.95	6.82		5.64		5.70
TOTAL WEIGHT (LBS) (INCLUDING PAYLOAD)	5220		4090		4160	8860		7320		7420
STAGE WEIGHT (LBS)	3670	860	2634	756	3460	6090	1470	4720	1300	6120
PROPELLANT WEIGHT (LBS)	3380	759	2389	652	3182	5658	1307	4323	1130	5662
DRY WEIGHT (LBS)*	290	101	245	104	278	432	163	397	170	458

*BASED ON FIRST ITERATION (λ')

TABLE 2 - STAGE SENSITIVITY FACTORS

		STAGES	S_1	S_2
ONE MAN	EARTH STORABLE	2	2.35	1.27
ASCENT	SPACE STORABLE	2 1	2.81 4.95	1.08 ----
TWO MAN	EARTH STORABLE	2	2.20	1.13
ASCENT	SPACE STORABLE	2 1	1.82 4.70	1.00 ----

TABLE 3 STAGE WEIGHT BREAKDOWN

VEHICLE	ONE MAN ASCENT (700 LBS)					TWO MAN ASCENT (1300 LBS)				
PROPELLANT	COMPOUND A/MMH		FLOX/METHANE			COMPOUND A/MMH		FLOX/METHANE		
STAGES	2		2		1	2		2		1
STAGE NUMBER	1	2	1	2	1	1	2	1	2	1
STAGE DRY WEIGHT (LBS)	284	102	262	113	302	448	159	395	159	468
WEIGHT SUMMARY										
STRUCTURE	(23.4)	(11.0)	(27.4)	(16.3)	(39.1)	(34.2)	(15.5)	(42.0)	(17.1)	(63.0)
TANKS	17.9	6.4	21.9	8.0	27.8	26.6	9.6	34.4	10.8	44.0
INTERSTAGE	4	2	4	2	5	5	3	5	3	9
THRUST STRUCTURE	1	0.3	1	0.3	2	2	0.6	2	0.6	3
INSULATION	0.5	0.3	0.5	4	4.3	0.6	0.3	0.6	7	7
BASE HEAT PROTECTION	-	2	-	2	-	-	2	-	2	-
PROPULSION SYSTEM	(136.9)	(50.6)	(121.0)	(49.6)	(130.8)	(225.4)	(81.1)	(186.3)	(76.5)	(206.6)
ENGINE	102	34.2	91.4	34.2	92.6	167	60	140	57.8	146
PRESSURIZATION SYSTEM	12.9	5.4	9.5	4.4	16.2	21.4	8.1	13.3	5.7	24.6
RCS	5	3	5	3	6	8	3	7	3	9
ENGINE GIMBAL	10	6	9	6	9	16	7	13	7	10
FILL, FEED, DRAIN & VENT SYSTEMS	7	2	6	2	7	13	3	[illegible]	3	13
EQUIPMENT & INSTRUMENTATION	(37)	(12)	(33)	(14)	(39)	(53)	(18)	(49)	(19)	(58)
CONTROL SYSTEM ELECTRONICS, ECS, MEASURING & TELEMETRY SYSTEMS CONTINGENCY (10% OF DRY WEIGHT)	32	9	28	11	33	44	14	42	15	50
SEPARATION SYSTEM	5	3	5	3	6		4	7	4	8
CONTINGENCY & REDUNDANCY ALLOWANCES (20% OF DRY WEIGHT)	64	18	56	22	66	88	28	84	30	100
RESIDUALS & RESERVE PROPELLANT	18	7	20	8	21	41	14	31	13	32
PROPELLANTS										
RCS PROPELLANT	5	3	5	3	6	8	3	7	3	9
USEABLE PROPELLANT	3215	765	2548	701	3453	5852	1271	4332	1102	5799
[illegible]	.920	.880	.900	.870	.910	.930	.890	.910	.880	.920
[illegible]	.919	.883	.907	.861	.920	.929	.889	.916	.874	.925
INSULATION WEIGHT ON DESCENT STAGE			51		43			92		81

TABLE 4 STAGE DIMENSIONS (EXCLUDING MANNED CAPSULE)

	EARTH STORABLE		SPACE STORABLE			
	ONE MAN	TWO MEN	ONE MAN		TWO MEN	
STAGES	2	2	2	1	2	1
LENGTH (IN)	133	147	136	108	139	108
LENGTH EXCLUDING ENGINE BELL EXTENSION (IN)	83	92	90	67	94	70
CROSS SECTION (IN)	64 x 30	66 DIA	64 x 30	66 DIA	64 x 30	66 DIA

TABLE 5 - PROPELLANT CHARACTERISTIC FOR SIZING TANKAGE

STAGE	COMPOUND A/MHF-5	COMMENTS	FLOX/METHANE	COMMENTS
MIXTURE RATIO	2.50	P_c = 1000 PSI	5.75 (T [illegible] 5000 LBF) 5.00 (T [illegible] 5000 LBF)	MR CHANGE DUE TO NON REGENERATIVE COOLING AT LOW THRUST. P_c = 1 000 PSI
FUEL DENSITY	62.8 LBS/FT^3	530 °R	27.0 LBS/FT^3	180 °R
OXIDIZER DENSITY	96.0 LBS/FT^3	530 °R	89.2 LBS/FT^3	180 °R
MAXIMUM TEMPERATURE	540 °R	EARTH STORAGE CONDITION	187 °R	EARTH STORAGE (BOILING POINT FLOX AT 67 PSIA)
MINUMUM TEMPERATURE	390°R	MHF-5 FREEZING POINT	163°R	METHANE FREEZING POINT
MAXIMUM/MINUMUM FUEL VAPOR PRESSURE	20 PSIA/> 1 PSIA	540°R/360°R	7 PSIA/> 1 PSIA	187°R/163°R
MAXIMUM/MINUMUM OXIDIZER VAPOR PRESSURE	56 PSIA/6 PSIA	540°R/360°R	67 PSIA/35 PSIA	187°R/163°R
FUEL ULLAGE	5%	ρ @ T_{MAX}	10%	ρ @ T_{MAX}
OXIDIZER ULLAGE	5%	ρ @ T_{MAX}	10%	ρ @ T_{MAX}

TABLE 6 - FUEL AND OXIDIZER TANK DESIGN PRESSURES

VEHICLE		COMPOUND A/MHF-5								FLOX/METHANE											
SIZE		1 MAN				2 MEN				1 MAN						2 MEN					
NUMBER OF STAGES		2				2				2				1		2				1	
CONTAINER		STAGE 1		STAGE 2		STAGE 1		STAGE 2		STAGE 1		STAGE 2		STAGE 1		STAGE 1		STAGE 2		STAGE 1	
TANK		FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX	FUEL	OX
VOLUME. FT (2 TANKS)		15.2	25.0	3.4	5.5	28.1	46.0	6.2	10.1	15.7	26.9	5.0	7.3	21.2	36.4	26.8	45.8	7.8	11.3	35.8	60.0
PRESSURE (PSI)	MINIMUM GAGE LIMIT	76	66	125	106	61	52	102	86	75	62	109	97	65	56	62	53	94	86	57	48
	INITIAL	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15
	VAPOR/STORAGE	16	54	16	54	13	48	13	48	7	67	7	67	7	67	7	67	7	67	7	67
	VAPOR/OPERATION	2	13	2	13	2	13	2	13	9	50	9	50	9	50	9	50	9	50	9	50
	NPSH	6	10	6	10	6	10	6	10	6	10	6	10	6	10	6	10	6	10	6	10
	ACCELERATION	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3
	LINE LOSSES	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4
	MAX. DUTY CYCLE (MDC)	16	54	16	54	+15 - 2	48	+15 - 2	48	22 -8	67	22 -8	67	22 -8	67	22 -8	67	22 -8	67	22 -8	67
	PROOF (1.1 MDC)	17	59	17	59	16	53	16	53	24	75	24	75	24	75	24	75	24	75	24	75
	DESIGN (YIELD) (1.1 PROOF)	19	65	19	65	18	58	18	58	27	81	27	81	27	81	27	81	27	81	27	81
TANK WEIGHT (LBS)		7.7*	10.2	2.7*	3.7	11.4*	15.2	4.2*	5.4	9.5*	12.4	4.0*	4.0	11.1*	16.7	13.3*	21.1	5.3*	5.5	16.4	27.6

*MINIMUM GAGE GOVERNING

TABLE 7 - ESTIMATED INSULATION REQUIREMENTS FOR EARTH STORABLE PROPELLANTS

VEHICLE	1 MAN		2 MAN	
STAGE	1	2	1	2
PROPELLANT WEIGHT (LBS) (2 TANKS)	3182	698	5860	1287
HEAT SINK CAPACITY/TANK/°F (BTU'S)	655	145	1210	266
TANK AREA (FT^2)	36	13	54	19
MAXIMUM THERMAL CONDUCTIVITY (BTU/ F/FT^2/HR)	.57	.35	.70	.44
DESIGN CONDUCTIVITY (BTU/ F/FT^2/HR)	.28	.17	.35	.22
DESIGN THICKNESS (IN)	.025	.041	.020	.032
INSULATION WEIGHT (LBS) (2 TANKS)	.45	.28	.54	.30

SPECIFIC HEATS: COMPOUND A .3 BTU/LB F
MHF-5 .7 BTU/LB F
MIXTURE RATIO 2.5

INSULATION: 3 LB/FT^3 POLYERETHANE FOAM h .007 BTU-IN/HR-FT^2-F°

TABLE 8 - SPACE STORABLE & INSULATION WEIGHT AND PERFORMANCE REQUIREMENTS

INSULATION CHARACTERISTICS	TWO STAGE		SINGLE STAGE		COMMENTS
	1 MAN	2 MAN	1 MAN	2 MAN	
CONTAINER AREA (FT^2)	174	315	144	274	
REQUIRED INSULATION PERFORMANCE (BTU/HR ^{o}F FT^2)	4.9×10^{-4}	4.5×10^{-4}	5.9×10^{-4}	5.2×10^{-4}	TWO WEEK STAYTIME ASSUMING 50% HEAT LEAK LOSS
WEIGHT (LBS)	51	92	43	81	CHARGE TO DESCENT STAGE EXCEPT FOR FORWARD CAPSULE INTERFACE

TABLE 9 - SPACE STORABLE AND INSULATION CHARACTERISTICS

RADIATION SHIELDS: .00024-IN-THICK ALUMINUM COATED POLYESTER

POLYURETHANE FOAM: 3 LB/FT^3 DENSITY 020 IN THICKNESS

NUMBER OF SHIELDS/IN: 151

PROPERTIES AT OPTIMUM DENSITY (VACUUM):

APPARENT THERMAL CONDUCTIVITY 1.0×10^{-4} BTU/IN/HR FT^2 ^{o}F

DENSITY 1.4 LBS/FT^3

PROPERTIES AT 15 PSI COMPRESSION:

APPARENT THERMAL CONDUCTIVITY 68×10^{-4} BTU/IN/HR FT^2 ^{o}F

DENSITY 3.0 LBS/FT^3

TABLE 10 - 42 LB PAYLOAD MSSR VELOCITY ACHIEVED BY UNMODIFIED MEM PROPULSION SYSTEM

CAPSULE \ PROPELLANT	EARTH STORABLE (2 STAGE)	SPACE STORABLE (ONE STAGE)
ONE MAN	35,900	31,200
TWO MAN	37,700	33,000

TABLE 11 - PAYLOAD TO 36,000 FPS AND 40,000 FPS WITH ADDED STAGE ON MEM

PROPELLANT	EARTH STORABLE (THREE STAGE)		SPACE STORABLE (TWO STAGE)	
CAPSULE \ VELOCITY	36,000 FPS	40,000 FPS	36,000 FPS	40,000 FPS
ONE MAN	77	27	102	46
TWO MAN	154	60	212	102